Future Directions of Supersonic Combustion Research: Air Force/NASA Workshop on Supersonic Combustion

Julian M. Tishkoff *
Air Force Office of Scientific Research
Bolling Air Force Base

J., Philip Drummond ✤
NASA Langley Research Center

Tim Edwards ❖
Abdollah S. Nejad ❖
Aero Propulsion and Power Directorate
Wright-Patterson Air Force Base

Abstract

The Air Force Office of Scientific Research, the Air Force Wright Laboratory Aero Propulsion and Power Directorate, and the NASA Langley Research Center held a joint supersonic combustion workshop on 14-16 May 1996. The intent of this meeting was to: (1) examine the current state-of-the-art in hydrocarbon and/or hydrogen fueled scramjet research; (2) define the future direction and needs of basic research in support of scramjet technology; and (3) when appropriate, help transition basic research findings to solve the needs of developmental engineering programs in the area of supersonic combustion and fuels. A series of topical sessions were planned. Opening presentations were designed to focus and encourage group discussion and scientific exchange. The last half-day of the workshop was set aside for group discussion of the issues that were raised during the meeting for defining future research opportunities and directions. The following text attempts to summarize the discussions that took place at the workshop.

* Program Manager, Associate Fellow, AIAA
✤ Senior Research Scientist, Associate Fellow, AIAA
❖ Senior Research Scientist, AIAA Member

Nomenclature

A	area
a	speed of sound
C_f	skin friction coefficient
D_1	Damkohler first number, L/ut_c
D_2	Damkohler second number, $\eta_c \Delta h_c / H_t$
E_a	activation energy
e_p	flow distortion
H_t	total flow enthalpy
h_t	specific enthalpy
L	combustor length
M	Mach number
Mc	convective Mach number, $(U_2-U_1)/(a_1+a_2)$
n	overall reaction order
P	pressure
q	dynamic pressure, $1/2(\rho u^2)$
R^0	universal gas constant
r	velocity ratio, U_2/U_1
s	density ratio, ρ_2/ρ_1
T	temperature

t_c characteristic combustion time
U, u flow velocity
w laminar burning rate
Z altitude
Δh_c heat of combustion
η_c combustion efficiency
ρ density

Subscripts

0 free stream condition
1, 2 stream 1, stream 2
4 isolator entrance condition
ad adiabatic flame condition
avg average value
max maximum value

Introduction

This paper summarizes the discussions held at an Air Force/NASA Workshop on Supersonic Combustion, in Newport News, Virginia on May 14-16, 1996. The purposes of the workshop were: (1) to review current design, performance, and testing practices for scramjets -- supersonic combustion ramjets used in high-speed airbreathing propulsion systems; and (2) to investigate the application of novel analytical methods, including experimental, theoretical, and computational approaches, to improve scramjet designs.

Recent programs for developing high-speed aerospace vehicles that utilize airbreathing propulsion provided the motivation for this workshop. Many of these programs were discussed at the recent AIAA 7th International Space Planes and Hypersonics Systems and Technologies Conference held in Norfolk, Virginia on November 18-22, 1996. Despite a high level of activity and financial investment in scramjet development for high-speed flight, no operational example of a scramjet currently exists. The cancellation of the United States National Aero Space Plane (NASP) program reflects the difficulties in developing this mode of propulsion successfully.

The intention of the organizers of the workshop was to provide a unique forum in which the developers and testers of propulsion technology could interact directly with members of the research community. The workshop was organized to intersperse formal presentations with open discussion in order to find common ground between two professional activities that otherwise might not have opportunities for such direct contact. To facilitate these interactions and discussions, invitations to attend the workshop were extended to approximately sixty participants, as summarized in Table 1. These participants were invited because of their experience and records of accomplishments in areas of research and technology relevant to scramjet design and testing. The organizers recognized that many other scientists and engineers possess knowledge and capabilities appropriate to the workshop but believed that an excessively large number of participants would hinder the interactions. The presence or absence of any scientist or engineer in Table 1 therefore does not represent anyone’s opinion about the professional merits of participants versus non-participants.

The workshop was conducted over a 2-1/2 day period. The first two days were devoted to presentations and related discussions. The topics and presenters are listed in Table 2. The body of this paper will review these presentations. This paper also may contain some additional ideas and comments that the authors have assembled since the workshop was held, but the primary content reflects the presentations and related discussions at the workshop. On the last half day of the workshop an open discussion was conducted in which general suggestions were made for future activities. A brief summary of these suggestions is given in Appendix A.

Table 1. Workshop Invitees

Name			**Affiliation**
Mr.	Griffin	Anderson	NASA Langley Research Center
Dr.	Fred	Billig	Applied Physics Laboratory, Johns Hopkins University
Dr.	Garry	Brown	Princeton University
Dr.	Dennis	Bushnell	NASA Langley Research Center
Dr.	Harsha	Chelliah	University of Virginia
Dr.	S M	Correa	GE Research Center
Dr.	E. T.	Curran	WL/PO, Wright Laboratory
Dr.	Stephen	D'Alessio	Applied Physics Laboratory, Johns Hopkins University
Dr.	Paul	Dimotakis	California Institute of Technology.
Mr.	Glenn	Diskin	NASA Langley Research Center
Dr.	James F.	Driscoll	University of Michigan
Dr.	J. Philip	Drummond	NASA Langley Research Center
Dr.	Craig	Dutton	University of Illinois
Dr.	Raymond	Edelman	Rocketdyne
Dr.	Tim	Edwards	WL/POSF, Wright Laboratory
Dr.	Fokion N.	Egolfopoulos	University of Southern California
Dr.	John	Erdos	GASL
Dr.	G. M.	Faeth	University of Michigan
Dr.	Alan	Garscadden	WL/CA, Wright Laboratory
Dr.	Peyman	Givi	State Univ. of New York
Mr.	Edward S.	Gravlin	WL/POP(HyTech), Wright Laboratory
Mr.	Wayne	Guy	NASA Langley Research Center
Dr.	R K	Hanson	Stanford University
Dr.	William	Heiser	HQ USAF/DFAN Department of Aeronautics
Dr.	Casey	Jachimowski	NASA Langley Research Center
Dr.	Ajay	Kumar	NASA Langley Research Center
Dr.	C K	Law	Princeton University
Dr.	Ron	Lehrach	United Technologies Research Center
Dr.	Frank	Marble	California Institute of Technology
Dr.	Atul	Mathur	Rocketdyne Division, Rockwell International. Corporation
Mr.	Chuck	McClinton	NASA Langley Research Center
Mr.	Bob	Mercure	NASA Headquarters
Lt Col	Richard	Moore	WL/POP, Wright Laboratory
Dr.	Abdollah	Nejad	WL/POPT, Wright Laboratory

Dr.	G. B.	Northam	NASA Langley Research Center
Dr.	Elaine	Oran	US Naval Research Laboratory
Dr.	Gerald	Pellett	NASA Langley Research Center
Dr.	S B	Pope	Cornell University
Dr.	David	Pratt	University of Washington
Dr.	David	Riggins	University of Missouri
Mr.	Kenneth	Rock	NASA Langley Research Center
Dr.	Clay	Rogers	NASA Langley Research Center
Dr.	Klaus	Schadow	Naval Air Warfare Center
Dr.	Joseph A.	Schetz	Virginia Polytechnic Inst. and State University
Dr.	Munir	Sindir	Rocketdyne Division, Rockwell International Corporation
Dr.	Mike	Smith	NASA Langley Research Center
Dr.	Louis	Spadaccini	United Technologies Research Center
Dr.	Scott	Thomas	NASA Lewis Research Center
Mr.	Michael	Thompson	Applied Physics Laboratory, Johns Hopkins University
Dr.	Julian M.	Tishkoff	AFOSR/NA
Mr.	Carl	Trexler	NASA Langley Research Center
Dr.	David	Van Wie	Applied Physics Laboratory, Johns Hopkins University
Mr.	Randy	Voland	NASA Langley Research Center
Dr.	Robert W.	Walters	AeroSoft, Inc.
Dr.	P J	Waltrup	Applied Physics Laboratory, Johns Hopkins University
Dr.	James	Weber	WL/POP
Dr.	Al	Wieting	NASA Langley Research Center
Dr.	Michael	Winter	United Technologies Research Center

Table 2. Workshop Agenda

1. Engine Design Issues (May 14, Morning Session)

Speakers:

Dr. Fred Billig, Johns Hopkins University, Applied Physics Laboratory
Mr. Chuck McClinton, NASA Langley Research Center
Lt Col Richard Moore, Wright Laboratory
Professor David Pratt, University of Washington

2. Ground Based Testing (May 14, Afternoon Session)

Speakers:

Mr. Michael Thompson, Johns Hopkins University, Applied Physics Laboratory
Mr. Randy Voland, NASA Langley Research Center

3. Fuels and Fuel Systems (May 15, Morning Session)

Speakers:

Dr. Tim Edwards, Wright Laboratory
Dr. Lou Spadaccini, United Technologies Research Center
Mr. Chuck McClinton, NASA Langley Research Center

4. Injection and Mixing (May 15, Morning Session)

Speakers:

Dr. Abdi Nejad, Wright Laboratory
Professor Garry Brown, Princeton University
Professor Paul Dimotakis, California Institute of Technology

5. Combustion Chemistry (May 15, Afternoon Session)

Speakers:

Professor Ed Law, Princeton University
Professor Harsha Chelliah, University of Virginia

6. Diagnostics and Simulation of High-Speed Flows (May 15, Afternoon Session)

Speakers:

Dr. Michael Winter, United Technologies Research Center
Dr. Munir Sinder, Rocketdyne

Engine Design Issues

The workshop began with a review of current practices for designing scramjet engines. Practical system issues such as mission requirements, integration of the inlet/isolator, combustor, nozzle, airframe, fuel system specification, and cooling concepts were addressed. The objective of this session was to discuss global design challenges associated with both cryogenic and hydrocarbon-fueled scramjets with the intent of identifying basic research opportunities to impact scramjet technology needs. However, at the time of the workshop, the Air Force had already defined a national program to develop technologies required for the development of a fixed geometry scramjet engine capable of operation over Mach 4 - 8 flight regime using conventional JP-based hydrocarbon fuels. Therefore, the majority of the discussion centered around technical challenges associated with the development of tactical missiles using storable fuels capable of acceleration from Mach 4 and cruise at Mach 8.

For this discussion, high speed vehicles were divided into the following two classes: a) aircraft or man-rated; b) expendable. The choice of high speed propulsion system (airbreathing, and rocket) hinges on many design and mission requirements. Factors such as size, weight, design complexity, maintainability, longevity, storability, production and life cycle costs, and logistic supportability were identified to be just as important as the performance characteristics (speed, range, and efficiency) of the hypersonic vehicle. Billig [1] listed some of the characteristics of hypersonic air-breathing vehicles, see Table 3. It is interesting to note that the combustor length remains virtually constant at 2-6 ft for the three classes of hypersonic vehicles, suggesting that supersonic combustion processes are inherently mixing-limited. The trade-off strategy to attain high combustion efficiency is much more complex in supersonic combustors, where shear losses can drastically reduce engine performance. Simply adding combustor length for optimization of mixing/combustion efficiency is usually not the prudent engineering solution.

The choice of air-breathing ramjet engine cycles depends on the flight Mach number. For example, at lower flight Mach numbers ($M < 5 - 6$) the subsonic integral rocket-ramjet is the preferred cycle. At high Mach numbers ($M > 6 - 7$) the scramjet cycle is the preferred mode of operation. However, a tactical missile -- an expendable, low cost, low weight, and therefore fixed geometry flow path design capable of operating at high flight Mach numbers $M > 6.5$ using conventional storable liquid hydrocarbon -- must operate as a ramjet at low flight speeds and as a scramjet at hypersonic speeds. Fortunately, if adequate combustor-inlet isolation is provided, the scramjet will function in a subsonic combustion mode at low Mach numbers with slightly lower efficiency than that of a conventional ramjet. However, a hydrocarbon-fueled scramjet designed to operate efficiently at Mach 7 - 8 using a fixed geometry flow path has not been shown to operate efficiently at Mach 4 flight conditions without resorting to use of massive auxiliary piloting [2], or without the use of large amounts of stored reactive oxidizer, e.g., chlorine trifluoride [3]. An interesting example of a massively piloted scramjet concept is the Dual Combustor Ramjet (DCR) which was designed and tested at Johns Hopkins University, Applied Physics Laboratory, and is schematically shown in Figure 1. This is an axisymmetric design in which the forebody serves as the initial compression surface of the supersonic inlet. In this concept, the incoming flow is divided into eight segments at the cowl lip. Four smaller inlets supply air to a subsonic dump combustor. They operate supercritically (the normal shock is swallowed) to avoid the interaction of the normal shock with the flow entering the larger inlets that feed the supersonic combustor. In order to provide stable combustor operation over a wide range of flight Mach numbers, the flow passages to the subsonic combustor have an increasing cross sectional area in the streamwise direction. The major portion of the air is captured by the four larger inlets and the external cowl compression surface and turned supersonically inward toward the engine axis. Captured flow is spread radially to form an annulus of supersonic flow that surrounds the outlet of the dump combustor. The aft sections of these supply ducts have slightly diverging flow passages in the streamwise direction, which effectively act as the combustor-inlet isolator. When the propulsion system is operating at a high equivalence ratio and/or at low flight Mach numbers, the isolator section can sustain a shock train with a pressure rise equivalent to that of normal shock. In this mode of operation the combustor inlet Mach number is less than one, and the mean Mach number at the combustor exit is either sonic or supersonic. At lower engine equivalence ratios and/or higher flight Mach numbers the isolator shock train pressure rise is equivalent to that of an oblique wave structure. With the inlet/isolator operating in the oblique shock mode, the mean flow Mach number throughout the scramjet is supersonic. This dual-mode engine operation has been discussed fully in the literature [4-6].

The issue of coupling combustor burner characteristics to vehicle cooling requirements is very important. The endothermicity of hydrocarbon fuels requires vehicle structural components to act as a heat exchanger/thermal cracking reactor. The composition of the cracked products depends on the time-temperature history of the cracking process throughout the vehicle structure. Changes in chemical composition or the state of the fuel directly affect burner operational characteristics; the time required for a radical pool to reach flammable conditions is linearly dependent on concentration, quadratically dependent on pressure, and exponentially dependent on the temperature. Therefore, precise control of the thermal cracking process is essential to the production of the desired fuel conversion (constituents) at the burner entry throughout the flight trajectory. However, the coupling of the heat exchanger/reactor to the combustor is not without its engineering challenges. Many tests of heat exchanger reactors have shown severe acoustic instability, leading to catastrophic failure. Tests of regeneratively cooled structures with endothermic fuels feeding a combustor have shown system instabilities between the two systems. The source of these acoustic instabilities may be the fact that hydrocarbon fuel remains near the thermodynamic critical point within the heat exchanger, where thermodynamic properties such as density, viscosity, latent heat, ratio of specific heats, and speed of sound show large variations with respect to small changes in temperature and pressure.

Mixing and heat release are significant engineering challenges in supersonic flows. However, when the engineer considers all aspects of the system design, mixing optimization, and/or combustion efficiency may not be the driving factors. Thus,

Mission	Flight Mach #	Propulsion System	Flow Path Geometry	Fuel	Flight Duration	Vehicle length (ft)
Tactical Missile	6 - 8	Dual Combustor Ramjet and/or Rocket	Fixed Geometry, passively cooled	Liquid HC, Slurry, Solid HC	10 -12 Minutes	Overall 5 - 15 Combustor 2 - 5 Nozzle 2 - 5
Trans-atmospheric Missiles	0 - 25	Dual mode Ramjet/Scramjet + many low speed options	Variable Geometry	Liquid H_2 Liquid O_2	20 - 30 Minutes 100 cycle	Overall 100- 200 Combustor 2 - 6 Nozzle 50 - 80
Hypersonic Cruise	0 - 8 0 - 15	Mach 6-8 Turboramjet M 15 scramjet	Variable Geometry, Actively Cooled	Mach 6-8, HC Mach 15, Liquid H_2	M = 6 - 8, 1 - 3 Hours M = 15 1 Hour	Overall 100- 200 Combustor 2 - 6 Nozzle 50 - 80

Table 3. General Characteristics of Hypersonic Vehicles

combustor and isolator lengths may not dictate the internal duct length. Since the internal drag can reduce the performance of a scramjet engine significantly, combustor designs with large surface areas should be avoided. Furthermore, the designers are usually careful in using intrusive injectors. Aside from the severe cooling requirements, the base and wave drag of many hyper-mixers render them ineffective in a practical device. Therefore, one must optimize and balance system overall performance, (i.e., maximizing net positive thrust), at the expense of not achieving complete mixing.

To develop a scramjet, designers require a design strategy. The following process was proposed: (1) start with a conceptual vehicle design; (2) optimize the design by sensitivity analysis; (3) select inlet(s), and conduct inlet tests, preferably in conjunction with the isolator, combustor, and injector components; (4) analyze the experimental data to update the cycle analysis codes to assess the performance potential of the scramjet design; (5) optimize the combustor/injector design concept. In order to implement this design strategy, accurate models for predicting jet penetration, mixing, combustion, heat transfer, and combustor-inlet interaction are required. To develop such models, research efforts must be ongoing for better understanding of the physics of supersonic combustion to evaluate and update the empirical design models used by the engineers. Free jet, semi-free jet, and direct connect tests must be conducted in sufficient detail to allow meaningful assessment of the performance and operational characteristics of the design and generation of benchmark data to aid with the development and validation of the analytical tools.

Ground Testing

The objective of this session was to introduce and discuss the state of testing and measurement technology used for assessment of scramjet performance in ground based facilities. The speakers outlined test procedures, instrumentation and measurement accuracy requirements, analytical modeling of the aerothermochemical processes, and error analysis procedures used for performance testing of the scramjet flow path.

Conventional ramjets and scramjets designed for Mach 6 - 8 flight push the limit of long duration (~ seconds to minutes) ground test direct-connect or free-jet test facilities.

Higher speed flight conditions (M >10) are simulated in pulsed facilities that can generate flight enthalpies in excess of M = 15 conditions, but only for a few milliseconds. In this session most of the discussion centered around testing scramjets in direct-connect and free-jet facilities. Figure 2 is a schematic illustration of a direct connect test facility. These facilities are relatively straightforward and are composed of the following key elements: (1) a high pressure air source; (2) an air heater (vitiated/arc-heated/pebble-bed/gas fired heat exchanger) for proper simulation of flight enthalpy; (3) a facility nozzle for proper simulation of combustor/isolator inlet Mach number in direct-connect tests or flight Mach number in free-jet tests; (4) a combustor and/or scramjet test article; (5) a load measuring system for thrust measurement; and (6) a steam calorimeter for estimation of combustion efficiency. Typical scramjet combustor entrance properties [7] are depicted in Table 4. In theory, it is desirable to duplicate or match these properties as closely as possible. However, practical requirements -- such as: power generation; fabrication of hardware to sustain the pressure; and facility and model cooling requirements for testing at flight enthalpy, which increase linearly with facility size (mass flow rate) and quadratically with flight Mach number-- may prevent duplication of all flight parameters. Anderson et al. [8] defined pressure, temperature, velocity, gas composition, and characteristic length scale as the primitive variables that describe the scramjet flowfield. Voland and Rock [9] have pointed out that, since complete duplication of the flight parameters in ground test facilities may not be possible, then one must identify parameters that impact the physical processes of supersonic combustion. It is generally agreed that these parameters are: flight Mach number, total enthalpy, Reynolds number, Stanton number, Damkohler first and second numbers, and the wall enthalpy ratio. Voland and Rock observed that the process of matching flight total enthalpy and Mach number allows proper simulation of the Damkohler second number D_2 -- the kinetic-to-thermal energy ratio. If the flight dynamic pressure is not matched due to power requirements or facility constraints and mass flow limitations forces, testing a smaller scale engine becomes necessary. Then the Damkohler first number D_1 -- the ratio of flow residence time to chemical reaction time -- is not simulated properly. If combustion is kinetically limited, then ignition delay characteristics of the fuel and the reaction times become a critical issue, and proper simulation of D_1 becomes critical. However, if the combustion is mixing limited, proper simulation of D_1 is not an issue. Dynamic pressure and geometric scaling also affect the ratio of the inertial to viscous forces (Reynolds number). Recall that the Reynolds number was identified as an important parameter to match in ground testing of engines. When scaling reduces the flow path size excessively, then one should question the extrapolation of the results due to mixing, shock-boundary layer interaction, boundary layer thickness, injector nozzle discharge coefficient, etc.

With few exceptions, instrumentation in these facilities is rather conventional and is limited to electromechanical devices for measuring pressure, temperature, gas composition, thrust, and combustion efficiency. Complexity and safety requirements compound the difficulty of incorporating advanced laser-based diagnostic techniques. Most often, steam calorimetry is used in long duration test facilities to quantify the amount of energy release, hence combustion efficiency. Several

	Free Stream Conditions						Isolator entrance Conditions					
Mo	Z_o (Kft)	Po (psia)	T_o (^oR)	U_o (ft/sec)	q_o (lbf/ft2)	h_{to} (BTU/lbm)	M_4	A_o/A_4	P_4/P_o	P_4 (psia)	T_4 (^oR)	U_4 (ft/sec)
3	47.95	1.868	390	2904	1694	133.3	1.529	2.86	7.8	14.51	744	2034
4	57.48	1.183	390	3872	1910	264.3	1.945	4.91	15.7	18.57	930	2885
5	65.72	0.7978	390	4840	2011	432.8	2.363	6.92	24.9	19.86	1102	3799
6	73.30	0.5569	394	5839	2020	646.7	2.767	8.91	35.3	19.65	1279	4770
7	80.07	0.4049	397.7	6844	2000	902.3	3.143	10.85	47.0	19.03	1451	5757
10	95.50	0.1984	406.1	9879	2000	1918.2	4.143	16.49	89.6	17.78	1958	8744
15	114.25	0.0857	424.8	15155	1945	4561.0	5.502	25.23	185.9	15.94	2880	13908
20	137.76	0.0319	460.8	21040	1287	8824.3	6.650	33.11	313.6	10.02	4074	19468
26.9	178.21	0.0067	480.5	28865	425.8	16629.8	7.688	40.12	472.9	3.20	5187	27205

Table 4. Typical ramjet/scramjet freestream and combustor inlet conditions

Measurement	η_c	C_f
Static Pressure	Good	Fair
Temperature	Good	Poor
Water Concentration	Good	Very Poor
Total Pressure	Poor	Good
Velocity	Very Poor	Very Poor

Table 5. Measurement Sensitivity

accurate measurements must be made to account for a proper energy balance from the heater to the calorimeter exit plane. These include: temperatures and flow rates of air, make up oxygen, fuel (heater and combustor), quench water, total temperature at the exit plane of the calorimeter, and heat loss through the facility nozzle and combustor walls. In this technique, water is injected downstream of the combustor exit plane to rapidly quench chemical reactions. The precision of the total temperature measurement at the calorimeter exit plane significantly impacts the analysis and the results. Stevens and Thompson [10] schematically illustrate the procedures used for analysis of an arc-heated facility, Figure 3. They also point out that various issues, such as precise determination of the heater stagnation condition, facility nozzle effective flow rate and discharge coefficient, combustor entrance and exit conditions, and calorimeter exit conditions are extremely important for precise estimation of scramjet combustion efficiency using a steam calorimeter.

In general it is recommended that, in addition to steam calorimetry, other measurements such as thrust, combustor pressure distribution, skin friction, Pitot pressure, and gas sampling should also be attempted. Table 5 shows the relative accuracy of the derived combustion efficiency (η_c) and the skin friction coefficient (C_f) as functions of measured parameters.

Fuels and Fuel Systems

"Fuel is becoming the integrating factor of the complete {high-speed vehicle} system" --E. T. Curran in [11]. "The problem is, we don't know how to make the scramjet combustor work efficiently using conventional fuels at low flight speeds corresponding to end-of-boost" -- F. Billig at this workshop.

There have been many recent workshops [11-13] and books [14, 15] in the supersonic combustion area that included discussions of fuels issues. A general consensus is: storable JP-type hydrocarbon fuels can be used up to Mach 6-8, although the upper end of this range will be a significant technical challenge that will require chemically reactive "endothermic" fuels. Lou Spadaccini of United Technologies Research Center briefed the workshop on endothermic fuels [16]. Liquid methane could be used to somewhat higher Mach numbers, but speeds in excess of about Mach 10 will require liquid hydrogen.

Air Force perspective

With the demise of NASP, the Air Force (AF) has focused its high-speed propulsion effort on storable hydrocarbon-fueled vehicles. Storable-fueled hypersonics is viewed as an important technology for the AF for various future missions [17]. However, hydrocarbon fuels have significant shortcomings in supersonic combustion when compared to hydrogen, notably relatively long ignition delays and limited cooling capability [12, 13]. One issue that needs to be addressed in practical engine design is the transition of the fuel injection and combustion processes that occur as the fuel temperature rises in the vehicle cooling passages. Early in the flight, cooling requirements are minimal, and the fuel is injected in a liquid phase. As the flight progresses and the flight speed increases, fuel may be heated to be well above its thermodynamic critical point. In both advanced gas turbines and scramjet engines, the fuel may be partially reacted (cracked or dehydrogenated) through its use as a coolant before reaching the combustor. It is of significant AF interest to determine the effect of this change in fuel character on the combustion process. It is anticipated that this partially reacted fuel will burn as well as, say, ethylene, with some claims that the combustion properties (such as ignition delay or reactivity) may approach or exceed that of hydrogen. Appropriate questions that need to be addressed are: (a) will the ignition delay of a partially cracked or dehydrogenated fuel under engine conditions approach that of (e.g.) ethylene or even hydrogen; and (b) how will the combustion efficiency/reactivity of a fuel change as it is heated and is partially reacted in the fuel system. The first step in kerosene-range hydrocarbon fuel combustion is often cracking of the C_{12}-level molecules to C_1-C_3 species. How will the combustion process be affected if these cracking reactions occur "upstream" of the combustor?

The use of fuel as a coolant in advanced engines can lead to thermal and catalytic reactions in the fuel, yielding H_2, CH_4, C_2H_4, C_2H_6, etc. [16, 18-21]. As these partially reacted, hot (e.g., 1200 ^{0}F/650 ^{0}C) fuels are injected into a gas turbine or scramjet combustor, it is appropriate to consider how the hot, partially reacted state of the fuel might affect the combustion process. As the liquid fuel is heated at pressure, it becomes a supercritical fluid with significantly different physical properties, such as density and viscosity [29]. This could be expected to significantly change injection behavior [30]. As the fuel begins to react in the fuel system, chemical changes in the injected fluid could also affect the combustion process. For example, ignition is considered to be a "radical-poor" process [22], and ignition delay is affected by radicals present due to air vitiation [23]. Are there sufficient radicals present in the "reacted" fuel at ~ 1200 ^{0}F to reduce ignition delay in a similar manner? In some cases, the reacted fuel can contain large mole percent levels of H_2, especially for endothermic fuels such as methylcyclohexane that are dehydrogenated. Does this H_2 content improve the ignition delay? Note again that the relatively long ignition delay time of hydrocarbons relative to H_2 is a key limitation for hydrocarbon-fueled scramjets [12, 13]. There is evidence from shock tube tests that the ignition delay of hydrocarbons is reduced by the presence of hydrogen, but still is orders of magnitude larger than that for pure hydrogen [24]. The kinetics of combustion are also of interest. Does the reacted fuel burn in a manner similar to its measured stable constituents, or does the presence of (significant?) amounts of hydrocarbon radicals change the reactivity? Another factor affecting combustion is that significant fractions of hydrogen could be generated in the fuel fed to the combustor either by fuel dehydrogenation or by "steam reforming" a fraction of the fuel $\{C_xH_y + xH_2O \rightarrow xCO + (x+0.5y)H_2\}$. An issue that may be significant is the effect of coke particles or soot precursors in the reacted fuel on combustion. Coking is a significant issue for high temperature fuels [16, 18, 25], and some fuels may form aromatics as part of the cracking process. Supercritical fuel increases the solubility of coke precursors (oligomers) from catalysts [26, 27]. How will these fuel-borne aromatics, particulates, and oligomers affect soot formation (and thus radiative heat loads in the combustor and emissions)?

The consensus at the workshop appeared to be that the answers to most of these questions are not known. To obtain this information, it was suggested that the effects of changes in the fuel must be studied in a realistic simulation of the scramjet combustion process, i.e. one that represents the diffusive nature of the combustion. One sub-scale possibility is co-annular or opposed-jet burners [28] that would burn hot, partially reacted fuels. Premixed combustion devices appear to be inadequate to address the important issues.

NASA Perspective on Fuels

The NASA Langley Research Center (LaRC) has been examining both hydrogen and hydrocarbon-fueled hypersonic vehicles concepts, including dual-fueled (H_2 + HC) vehicles. Dual-fueled systems have advantages, as demonstrated by the dual-fueled Apollo missions. Chuck McClinton briefed the workshop on the status of LaRC's scramjet work in these areas. NASA studies have confirmed the Mach 7-8 limit for hydrocarbon-fueled vehicles. NASA work, as discussed above, has shown the ignition, combustion, and cooling difficulties of hydrocarbon fuels. Hydrogen is a much better scramjet fuel, except in the areas of volumetric fuel energy density and logistical supportability. Published NASA vehicle designs for both hydrocarbon-fueled [31] and hydrogen-fueled [32] vehicles were mentioned. NASA is supporting the Air Force HyTech program with analysis and modeling, although the primary focus of NASA/LaRC's work is flight tests of a H_2 dual-mode scramjet system.

Combustion Chemistry

This portion of the workshop addressed the identification of detailed chemical kinetic mechanisms for scramjet combustion and the reduction of those mechanisms to produce kinetic models for combustor design codes. Discussions were also directed at approaches to model turbulence-chemistry interactions.

The computational complexity of solving turbulent fluid transport equations provides a strong incentive for simplifying the description of chemical kinetics as much as possible in combustor design codes. The degree of success of such simplifications depends on the information that is required for each calculation. For example, equilibrium chemistry is adequate for calculations of non-optimum, non-critical global performance and has been used successfully for such applications as predicting overall energy release in internal combustion engines. However, the accuracy of simplified or reduced chemistry must be scrutinized carefully for other calculations.

An example of the limitations of simplified chemical kinetic models can be found in the calculation of laminar flame propagation using one-step global chemistry [33]. Equation 1 provides an Arrhenius expression to represent one-step model for the laminar burning rate w:

$$w \sim P^{n/2} \exp[-E_a/2R^0T_{ad}] \qquad (1)$$

where symbols are defined in the Nomenclature.

The simplest form of the one-step expression would have n as a constant. However, even if n were treated as a pressure-dependent variable, this expression can be shown to be deficient.

To test the validity of eq. (1) with n as a variable, Egolfopoulos and Law [33] measured the laminar flame propagation of methane-oxygen-nitrogen mixtures in a counterflow, twin-flame configuration. Figures 4-5 show the behavior of the laminar burning rate w and the overall reaction order n, respectively. According to eq. (1) w should exhibit a monatonic, exponential dependence on pressure. Figure 4 does not confirm this dependence. Figure 5 shows that the exponent n is always less than 2, which is incompatible with n as a constant. n also has considerable variation with pressure and even can assume negative values. Thus, eq. (1) is a poor estimator of laminar flame behavior.

The physicochemical basis for the deficiency of eq. (1) lies in the inability to account for the complexities of the competition between two-body chain branching reactions and multibody termination reactions in determining flame propagation. The presence of nonreactive third bodies to serve as collision sites in the termination reactions makes these reactions particularly sensitive to pressure.

If more complex reduced chemical mechanisms are needed, then how are they to be derived? The essential first step in producing reduced kinetic mechanisms is the identification of complete chemical reaction mechanisms for representative fuel combustion conditions. For hydrogen fuel, this process is straightforward. For example a complete chemical reaction mechanism for H_2-O_2-CO_2 can involve 13 species and 27 reaction steps. However, for hydrocarbon fuels, it is more complex and difficult. Even a simple methane-air mechanism can include 16 species with 40 reaction steps, while hydrocarbon-air combustion chemistry can involve 40 species with 100 reaction steps for more complex hydrocarbon species. In hypersonic applications, with fuel needed for cooling purposes, the identification of specific fuel components represents the initial challenge. For example, recent testing of endothermic fuels suggested that ethylene was a major product of endothermic catalytic reactions [16]. However, more recent results, as discussed by Edwards in this workshop, contradict this choice. Processes such a soot formation remain elusive because of their complexity.

A second obstacle to the measurement of complete reaction mechanisms is limitations in reactor and diagnostic capabilities. Kinetics must be measured under thermodynamic and fluid dynamic conditions that simulate high speed propulsion environments. Note that, measurement capability must be adequate for all critical species.

Two steps are generally used for simplifying chemical kinetic mechanisms: development of "starting" mechanisms and "reduced" mechanisms. The starting mechanism represents a subset of the detailed mechanism, obtained by elimination of elementary reactions to diminish the number of total species in the system by as much as 90%. Further simplifications of the starting mechanism may be achieved by the introduction of systematic "reductions" based on the chemical and flow time scales of the problem. Since calculation times ~ (number of species)^2, these simplifications can produce dramatic savings in computational time.

Two approaches have been identified to produce starting mechanisms:

1. "Systematic" approaches first use intuitive arguments to eliminate noncritical species and then use sensitivity analysis to reduce the number of reaction steps. Peters [34] introduced steady-state or partial equilibrium approximations to achieve such simplifications. This approach raises concerns that the results may be specific to the type of flame being calculated.

2 Automated procedures. This systematic approach produces mechanisms that span the full range of known experimental results and should not be unique to any individual experiments. Automated reaction procedures have been suggested by Lam [35], Chelliah [36], and Pope [37]. This approach has been applied to unsteady zero dimensional (homogeneous) systems but not as yet to combustion involving diffusive transport. Figure 6 [36] illustrates the application of this approach to predict heat release in a nonpremixed counterflow methane-air flame. This figure shows a comparison between a 16 species, 40 reaction step starting mechanism and two systematically reduced mechanisms, developed by introducing steady state approximations. A 31% representative saving in computational time may be expected from such reductions. Similar calculations are underway for oblique detonation wave combustion.

A strategy was suggested to implement automated reduction:

1. Obtain a detailed, comprehensive data base for C_7-C_{12} aliphatic fuels.

2. Select a surrogate fuel.

3. Derive [34-37] an appropriate reduced mechanism for the intended application.

Table lookup procedures were suggested in the workshop as an alternative to embedded solution of reduced chemical kinetic equations in combustion calculations. Lookup methods are linearly proportional to the number of species, as opposed to the quadratic dependence noted above for reduced mechanisms. The utility of lookup procedures depends on computational efficiency. Pope [37] recently has suggested novel methodology to accelerate table lookup.

Turbulence-chemistry interactions represent a major complication, coupling chemical kinetic behavior to fluid transport. Shear flows in supersonic combustion will produce strain rates, with a correspondingly large variation in scales affecting ignition, flame stability, and diffusion. Compressibility introduces additional complications, in which dilatation provides a wave source that impacts combustion.

Diagnostics

The focus of this portion of the workshop was on the application of current optical measurement techniques to scramjet research and testing. Eckbreth [38] describes the fundamental principles on which these techniques are based. Hanson [39] provides an overview of imaging methods in combustion flows.

Table 6 summarizes the techniques that were discussed. Each of the five techniques has been applied under actual or simulated propulsion system testing environments. However, with the exception of fuel plume imaging, none of them can be considered to be a standard in current testing facilities.

The benefits of the techniques in Table 6 can be appreciated by comparing them to the current state of capability for high speed propulsion system testing. Particularly at high flight Mach number conditions ($M > 10$), ground-based testing is limited to transient facilities such as shock tubes. In such facilities

TECHNIQUE	PARAMETERS MEASURED	ADVANTAGES	DISADVANTAGES	COMMENTS
Fuel Plume Imaging (Lorenz-Mie Scattering)	Plume Geometry, Mixing Efficiency	Strong Signal, Experience With Application	Need To Introduce μm Seed Particles, Behavior At Flame Front	Initial Difficulties With Seeding Overcome By Silica Dry Seeding Technique
Rayleigh Scattering	Plume Geometry, Temperature, Mixture Fraction	Simplicity, Multiparameter, Multidimensional Information	Lower Signal-Noise Ratios Than Lorenz-Mie Scattering, Background Interference	
Iodine Fluorescence	Time-Averaged (20 s) Velocity, Pressure, Temperature, Species Concentration	Multiple Parameter, Multidimensional Data	Errors in Interpreting Time-Averaged Parameters From Time-Averaged Data, Expense, Alignment	Has Provided Valuable Data for Scramjet Code Validation, But Limited To Nonreacting Flow Conditions
Coherent Anti-Stokes Raman Spectroscopy	Multiple Major Species, Temperature	Mature, Quantitative, Multi Parameter. Instrumentation Can Be Remote From Measurement	Single Point, Low Signal Strength	Has Been Demonstrated In Operational Testing Environments, Such As The Plume of an F100 Turbofan Engine
Planar Laser-Induced Fluorescence	Multiple Minor Species, Pressure Temperature, Velocity	Time-Resolved Multidimensional, Multiparameter Measurements	Expensive, Requires Careful Alignment, Although Less Than CARS. Difficult to Quantify	Attempts To Apply to Scramjet and Hypersonic Testing Have Met With Mixed Success

Table 6 Non-intrusive Diagnostics Measurement Techniques

time per test can be limited to milliseconds, with only a few tests per day. Thus, data acquisition becomes a primary factor in establishing the duration and cost of any testing program.

Methods currently used in scramjet testing include electromechanical devices such as thermocouples and pressure transducers for quantitative information, photographic and videographic image recording, spontaneous emission spectroscopy, and mechanical sampling. The electromechanical devices are point measurement techniques, so that an extensive array is needed to determine time-resolved spatial variations in temperature and pressure. These measurements are intrusive into the flowfield if they are mounted on probes. Otherwise they are restricted to surface characterization.

Image recording can be based on emitted light or on shadowgraph or schlieren approaches, which utilize a light source. Images recorded in this manner provide path-averaged, qualitative interpretations of flowfield behavior. Attempts have been made to expand schlieren capability by spectrally-resolved recording (color schlieren).

Sampling and spontaneous emissions spectroscopy have provided data on combustion chemistry. Sampling is a point measurement that is not temporally resolved. The extraction of the sample also can allow additional chemical reactions to occur in the sample, representing a source of error in the measurement. Spontaneous emission spectroscopy shares the path averaging limitations of image recording methods and requires assumptions regarding excited state population fractions.

As indicated in Table 6, recent advances in laser-based measurement techniques offer the hope of overcoming many of the limitations noted above for current measurements. However, the application of these methods to practical testing environments is in its infancy, and lessons learned thus far show that the application process generally will not be straightforward or easy.

In Table 6 the results for fuel plume imaging were based on seeding the injected fuel with silica (SiO_2) particles as scattering sites for light. As noted in the table, initial problems were encountered by approaches to create these particles through the reaction of silane with oxygen according to the following reaction mechanism:

$$H_2 + SiH_4 + 2O_2 \rightarrow H_2 + SiO_2(s) + 2H_2O + \text{heat}$$

These approaches produced some problems with nonuniform particle size and particle agglomeration. Furthermore, water vapor would then be present in the fuel as a consequence of this chemical reaction. Subsequently, an alternative approach in which uniform size silica particles were seeded directly into the fuel prior to injection removed this difficulty. For both approaches particle agglomeration and residues proved to be problems. Particle vaporization also was a concern, and the technique worked best for noncombusting tests, in which fuel was injected into nitrogen gas.

Experience with the optical methods of Table 6 has indicated some common areas of concern:

1. Windows. Optical access is a major design concern for a test apparatus. Proper design requires windows to be sufficiently durable and not to alter system behavior.

If cleaning or replacement is required, then ease in performing these maintenance functions should be incorporated into the design of the apparatus.

2. Noise and vibration. Noise should be considered in the general electromagnetic sense. Electrical noise from flow generating equipment and from instrumentation can be a significant interference to low amplitude signal generation. The Lorenz, Mie, and Rayleigh scattering techniques in Table 6 are examples of elastic scattering, in which the signal radiation is at the same wavelength as the incident radiation. Therefore, spectral filtering methods to reduce background radiation may be more difficult than with the laser-induced fluorescence and coherent anti-Stokes Raman measurements. Mechanical vibrations represent a serious challenge to optical alignment and the durability of optical components and lasers.

3. Extreme thermodynamic conditions. The high temperatures and pressures associated with combustion testing represent a hazard to both measuring equipment and human operators. In some previous tests measurement system design required a capability for remote adjustment and alignment while measurements were being performed.

4. Environmental and safety regulations. The safety requirements for propulsion test facilities and those for the operation of laser-based measurement instrumentation are not always directly compatible. Recent experience at one scramjet testing facility that was located outdoors near an airport necessitated enclosing laser beams to shield them from nearby aircraft. These considerations impose additional design requirements that are not directly related to the measurements.

The potential benefits of the methods listed in Table 6, as well as other methods currently under study, must be taken into consideration in future scramjet development programs. Although these methods generally have not been utilized sufficiently to make their application easy as yet, in some cases unique and extremely valuable data have been obtained that could not be measured by any other means. The rapid advancement of optical measurement technology, including such developments as fiber optics and diode lasers, should facilitate their adoption in the future. Furthermore, routine usage should simplify measurement practices, so the participation of Ph.D spectroscopists will not be required.

Simulation

The role of computational fluid dynamics (CFD) in the design of a hypersonic propulsion system was described by Sindir in this session. The application of computational techniques to major scramjet components, including the inlet/isolator, combustor, and nozzle, was first discussed. The relevant flow physics in each component was considered, followed by the current approaches for analyzing that flowfield. Deficiencies in the current approaches were then described, and new technology required to deal with these deficiencies were discussed. The experiments and data needed to validate the computational tools applied to each component were also discussed. Following the discussion of the analysis and design of individual engine components, modeling of the integrated flow path was considered.

CFD has several roles in the design of a hypersonic propulsion system. It primarily serves as an engineering tool for detailed design and analysis. In addition, results from CFD analyses provide input data for cycle decks and performance codes. Finally, CFD has several uses in engine test programs used to develop an engine concept. CFD is first used to guide the test setup and to determine the proper location for the placement of instrumentation in the engine. It has also proven to be an effective tool for determining the effects of the facility on testing, for example, the effects of contaminants in a combustion heated facility on an engine combustor test. During and following a test, CFD is useful to predict flowfield measurements as a complement to measured data.

The inlet/isolator of a scramjet engine supplies the combustor with a required quantity of air at a specified pressure, velocity, and flow uniformity. The physics of the flow in an inlet are characterized by:

1. Moderate strength shock waves
2. Shock-boundary layer interactions
3. Flow separation in unfavorable pressure gradients
4. Compressibility effects
5. Transition to turbulence
6. High leading edge thermal loads
7. Possible unstart

Computational analyses of inlets typically employ codes that solve the Euler equations or Euler iterated with the boundary layer equations for viscous effects for initial analyses. More detailed calculations utilize either the parabolized Navier-Stokes equations or the full Navier-Stokes equations if significant flow separation must be considered. All of the calculations typically solve the steady-state equations so that the simulations can be completed in reasonable times. Turbulence is modeled using either algebraic or two-equation turbulence models with empirical compressibility corrections and wall functions. Transition models are not currently being employed. Thermodynamic properties are generally determined by assuming that the inlet flow behaves as a perfect gas or equilibrium air. Calculations are conducted on fixed grids of 100,000 to 2,500,000 points in multizone domains. A limited degree of dynamic grid adaptation is employed when necessary. Typical run times range from a few minutes to 50 hours on a Cray C-90 computer.

A typical high-speed inlet calculation by Sindir is shown in Figure 7. The inlet shown in the figure utilizes side wall compression to achieve the desired outflow conditions into a combustor. The flow in the inlet is modeled using a full Navier-Stokes code with an algebraic turbulence model. The calculation is conducted on a grid of 240,000 points. Computed pressure contours are superimposed on the picture of the inlet. The plot in the figure shows a comparison between the computed wall pressures plotted as a function of downstream distance and measured data. The agreement between the computation and the measured data is excellent. Data away from the inlet walls is not available for comparison.

Based on the current state of the art for inlet calculations and the future technology needs, the following advancements are needed. More efficient parabolized and full steady-state Navier-Stokes codes with a factor of five increase in run time efficiency are needed. Significant improvements are also required for temporal Navier-Stokes codes for the analysis of unsteady inlet flowfields, including inlet unstart. Improvements should occur with algorithmic advancements, with one promising area being multigrid methods [40]. Continuing advancements in computer architectures will also enhance code speed. Improved methods for dynamic grid adaptation would also enhance the ability of computational algorithms to capture flowfield features. There is a serious need for the development of advanced transition and turbulence models. This is likely the most limiting area for accurate modeling of inlet flowfields. Promising work is now underway to develop new algebraic Reynolds stress turbulence models with governing equations that can be efficiently solved [41, 42]. For nonequilibrium flows, the differential Reynolds stress equations must be solved, however, and further work is necessary for this to be done more efficiently. Advances in large eddy simulation, with the development of subgrid scale models appropriate to high-speed compressible flow, may also allow this technique to be applied to inlet flows in the future [43]. Finally, work is needed to develop improved transition models for inlet flows, particularly with flows exhibiting adverse pressure gradients.

Experiments must also be conducted to provide code validation data for inlet flowfields. When these experiments are conducted, more extensive wall pressure measurements are required, along with detailed wall heat transfer and skin friction data. There should also be an accurate definition of the shock structure present in the inlet flow. Finally, in addition to the wall pressure measurements, in-stream measurements are critical for code validation. Initially, velocity profiles would be very useful. Pressure and temperature profiles are also needed. Measurements of these quantities in high-speed compressible flow are quite difficult, stretching the state-of-the-art in flow diagnostic techniques. To accurately measure these quantities in inlet flows, significant work will also be required to develop nonintrusive diagnostic techniques to collect the required validation data.

The flowfield in the combustor of a scramjet engine is characterized by much of the flow physics of the inlet, but it is further complicated by:

1. A wide range of flow velocities inhomogeneously distributed throughout the combustor.
2. Small and large scale vortical flows (for mixing).
3. Separated flows (for flameholding)
4. Complex mixing phenomena.
5. Finite rate chemical reaction (that may equilibrate).
6. High temperatures and heat fluxes
7. High degrees of anisotropy and nonequilibrium transfer of turbulence energy.
8. Interactions between turbulence and kinetics that affect chemical reactions and the turbulence field.

Computations of combustor flowfields typically employ codes that solve either the parabolized or full Navier-Stokes equations, depending upon the region of the combustor being modeled and the degree of flow separation and adverse pressure gradient being encountered. Steady-state methods are normally used with limited unsteady analyses for mixing studies or the analysis of combustion instabilities. Turbulence is again modeled using algebraic or two-equation models with empirical compressibility corrections and wall functions. There is a limited use of models to account for turbulence-chemistry interactions based on assumed probability density functions. Thermodynamic properties are determined utilizing perfect gas or, in some cases, real gas models. Chemical reaction is modeled with reduced reaction set finite rate models. For the hydrogen-air reactions occurring in a hydrogen fueled scramjet, a typical reaction mechanism includes nine chemical species and eighteen chemical reactions, although other mechanisms are employed as the case dictates [44]. Hydrocarbon-fueled scramjet concepts are modeled with more complex mechanisms that must be further reduced to allow practical computations. Calculations in each case are typically conducted on fixed structured grids of 200,000 to 2,500,000 points in multizone domains. Typical run times on a Cray C-90 computer range from 30 to over 300 hours.

The results of a calculation of the near-field of a transverse fuel injector design utilized in a scramjet combustor is shown in Figure 8 [45]. Conventional scramjets utilize streamwise fuel injection in the lower Mach number regime to produce the desired heat release schedule in the combustor. In the higher Mach number regime, some transverse injection is utilized to increase mixing in order to achieve the required heat release schedule with shorter combustor residence times. The flow near an aligned pair of transverse fuel injectors downstream of a rearward facing step is diagrammed in the Figure 8. In this study, air mixed with a small amount of iodine injected at Mach 1.35 is used to simulate the fuel. The iodine allows the injectant to be measured and tracked as it mixes with upstream air initially introduced at Mach 2. A comparison of the measured and computed mole fraction of injectant in a streamwise plane cutting through the center of the injectors is also shown in Figure 8. The agreement between the experimental data and the computed results is quite good.

Many of the future technology needs for combustor simulations follow from the needs for inlets described earlier, but many of the additional requirements will be more difficult to achieve. For combustor modeling, a factor of ten improvement in the efficiency of steady-state and temporal Navier-Stokes codes will be needed to carry out the required calculations with the necessary accuracy and design turn-around time. Multigrid methods again offer promise for significantly enhancing convergence rates, but the application of multigrid methods to reacting flows also results in additional challenges for success with the method [40]. Current research to apply multigrid methods to high speed reacting flows has resulted in a significant improvement in convergence rates over single grid methods. Dynamic grid adaptation will become even more important for capturing the complex flow structure in combustors, in particular the shock-expansion and vortical structure in the flow. Proper resolution of vortical flow requires very high resolution to conserve angular momentum. Again, there is a serious need for improved turbulence modeling in high speed reacting flows, both to model the turbulence field and to properly couple the effects of turbulence on chemical reaction and reaction on turbulence. Promising work is again taking place in this area using several approaches. Techniques using velocity-composition probability density functions have been successfully applied to incompressible reacting flows, and this work is now being extended [46], to model compressible reacting flows. Work is also underway [43] to apply large eddy simulation

(LES) techniques to compressible reacting flows. Subgrid scale models for the LES of these flows are currently being developed using methods previously applied for modeling the full range of flow scales. Finally, further work is needed to simplify the modeling of chemical reaction in combustor flowfields. Methods for systematically reducing the number of reactions in a full reaction mechanism are required to reduce the computational work [47]. A number of promising methods are under development. They were discussed in a previous section.

As with the modeling of inlet flowfields, experiments are also required to provide data for the validation of combustor codes. In addition to the data required for validating inlet modeling, combustor code validation will require extensive temperature and species concentration measurements, as well as the correlations of these quantities with each other and with velocity for validation of advanced turbulence models. Measurements of all of the required flow variables are more difficult to obtain in the reacting flow environment of a scramjet combustor. Significant work will again be required to develop nonintrusive diagnostic techniques suitable for making the required measurements.

The flowfield in the nozzle of a scramjet engine is characterized by much of the flow physics of the inlet and combustor, but additional requirements include the modeling of:

1. Strong aerodynamic and chemical non-uniformities.
2. Very high velocities and high temperatures.
3. Significant divergence and skin friction losses.
4. Changing thermochemical state.
5. Potential relaminarization of the flow.
6. Energy-bound chemical radicals that will not relax in the nozzle.
7. Excited vibrational states and their relaxation.

Computations of nozzle flowfields are usually conducted with Euler codes or Euler codes iterated with boundary layer calculations for initial engineering design studies, and with either parabolized or full Navier-Stokes codes for more detailed studies. Steady-state methods are normally employed. Turbulence is modeled by algebraic or two-equation models with empirical compressibility corrections and wall functions. Perfect gas or, when necessary, real gas models are used to determine thermodynamic properties. Chemical reaction is modeled with reduced kinetics models as utilized in the upstream combustor flow. Finite rate analyses are still required in the nozzle to assess the degree of reaction that continues to take place and to determine the extent of recombination reactions that add to the available thrust. Calculations for complete nozzles are typically carried out on structured grids of 100,000 to 500,000 nodes grouped in multizone domains. Typical run times range from 1 to 40 hours on a Cray C-90 computer.

The results of a simulation by Sindir to optimize nozzle performance are given in Figure 9. A parametric study is performed on a three-dimensional nozzle using a distribution of inflow profiles that are given in the figure. Profiles are characterized in terms of the flow distortion, given by $e_p = P_{avg}/P_{max}$. Mass and stream thrust are held constant for all of the profiles. Simulations using each profile are conducted using a 3D Euler code. The effects of the various flow profiles are characterized in terms of nozzle efficiency, thrust, and thrust vector angle. Plots of nozzle efficiency and thrust vector angle vs. the distortion parameter are also given in Figure 9. Clearly, nozzle performance is greatly affected by flow non-uniformity. Efficiency tends to increase when the distortion parameter becomes more negative with increasing pressure toward the cowl side of the engine. Therefore, high inflow distortion, oriented appropriately, can favorably affect nozzle performance.

Future technology needs for nozzle simulations, even though less demanding, follow very similar lines to the requirements for combustor simulations. A factor of five improvement in the efficiency of the steady-state Navier-Stokes codes is needed. Dynamic grid adaptation will also be useful for capturing shock structure and resolving possible wall separation due to shock-boundary layer interactions. There is a need for improved turbulence models for describing nozzle flows. Algebraic Reynolds stress turbulence models offer significant promise for describing these flowfields [41, 42]. The reduced kinetics models currently being applied to nozzle flows appear to be reasonably accurate, although some further work to improve the description of recombination may be warranted. Validation requirements for nozzle codes are similar to those required for combustor codes.

Injection and Mixing

The critical issues of fuel injection and mixing in a scramjet combustor were discussed in this session by Nejad, Brown, and Dimotakis. A number of key issues for efficient fuel injection, mixing and combustion were first considered. The shear/mixing layer flow was then discussed to provide a mechanism for a better understanding of the fundamental physics of fuel-air mixing and combustion. A number of conventional fuel injection strategies were then described followed by several new less conventional techniques. Finally, an appraisal of these injection strategies were made.

There are several key issues that must be considered in the design of an acceptable fuel injector. Of particular importance are the total pressure losses created by the injector and the injection processes, that must be minimized since they reduce the thrust of the engine. The injector design also must produce rapid mixing and combustion of the fuel and air. Rapid mixing and combustion allow the combustor length and weight to be minimized, and they provide the heat release for conversion to thrust by the engine nozzle. The fuel injector distribution in the engine also should result in as uniform a combustor profile as possible entering the nozzle so as to produce an efficient nozzle expansion process. At moderate flight Mach numbers, up to Mach 10, fuel injection may have a normal component into the flow from the inlet, but at higher Mach numbers, the injection must be nearly axial since the fuel momentum provides a significant portion of the engine thrust. Intrusive injection devices can provide good fuel dispersal into the surrounding air, but they require active cooling of the injector structure. The injector design and the flow disturbances produced by injection also should provide a region for flameholding, resulting in a stable piloting source for downstream ignition of the fuel. The injector cannot result in too severe a local flow disturbance, that could result in locally high wall static pressures and temperatures, leading to increased frictional losses and strict wall cooling requirements.

Compressible shear/mixing layers and jets provide a good model problem for studying the physical processes occurring in high-speed mixing and combustion in a scramjet. Mixing layers are characterized by large-scale eddies that form due to the high shear that is present between the fuel and air streams. These eddies entrain fuel and air into the mixing region. Stretching occurs in the interfacial region between the fluids leading to increased surface area and locally steep concentration gradients. Molecular diffusion then occurs across the strained interfaces. There has been a significant amount of experimental and numerical research to study mixing layer and jet flows [48-56]. For the same velocity and density ratios between fuel and air, increased compressibility, to the levels present in a scramjet, results in reduced mixing layer growth rates and reduced mixing. The level of compressibility in a mixing layer with fuel stream 1 and air stream 2 can be approximately characterized by the velocity ratio, $r = U_2/U_1$, the density ratio, $s = \rho_2/\rho_1$, and the convective Mach number, $M_c = (U_2-U_1)/(a_1 + a_2)$ where a is the speed of sound. Increased compressibility reorganizes the turbulence field and modifies the development of turbulent structures. The resulting suppressed transverse Reynolds normal stresses seem to result in reduced momentum transport. In addition, the primary Reynolds shear stresses responsible for mixing layer growth rate also are reduced. The primary mixing layer instability becomes three-dimensional with a convective Mach number above 0.5, reducing the growth of the large scale eddies. Finally, the turbulent eddies become skewed, flat, and less organized as compressibility increases. All of these effects combine to reduce the growth rate of the mixing layer and the overall level of mixing that is achieved.

Several phenomena result in the reduction of mixing with increasing flow velocity, including velocity differential between fuel and air, and compressibility. Potentially, the existence of both high and low growth and mixing rates are possible, and the engine designer with an understanding of the flow physics controlling these phenomena can advantageously use these effects. The shock and expansion wave structure in and about the mixing layer can interact with the turbulence field to affect mixing layer growth [48]. Shock and expansion waves interacting with the layer result from the engine internal structure. Experiments have shown that the shocks that would result from wall and strut compressions appear to enhance the growth of the two-dimensional eddy structure (rollers) of a mixing layer. This effect is most pronounced when the duct height in the experiment and the shear layer width become comparable. Waves may be produced by the mixing layer itself under appropriate conditions. Localized shocks (often termed shocklets) occur within the mixing layer when the accelerating flow over an eddy becomes supersonic even when the surrounding flow is subsonic. When the overall flow is supersonic, the eddy shocklets will extend as shocks into the flow beyond the individual eddies. These shocklets can retard eddy growth due to increased localized pressure around the eddy.

The growth of a mixing layer produces a displacement effect on the surrounding flow field. This displacement in confined flow produces pressure gradients that can affect the later development of the mixing layer, typically retarding growth. When chemical reaction occurs in a mixing layer, resulting in heat release, the growth of the mixing layer is retarded in both subsonic and supersonic flow [48, 49]. The effect of heat release can also vary spatially as a function of the local stoichiometry and chemical reaction. Dimotakis noted that the retarded growth in both instances can be reversed, however, by allowing the bounding wall to diverge relative to the initial wall angles where retarded growth was noted [50].

Several options are available for injecting fuel and enhancing the mixing of the fuel and air in high speed flows typical of those found in a scramjet combustor. Nejad discussed the two traditional approaches for injecting fuel include injection from the combustor walls and in-stream injection from struts. The simplest approach for wall injection involves the transverse injection of the fuel from wall orifices. Transverse injectors offer relatively rapid near-field mixing and good fuel penetration. Penetration of the fuel stream into the crossflow is governed by the jet-to-freestream momentum flux ratio. The fuel jet interacts strongly with the crossflow, producing a bow shock and a localized highly three-dimensional flow field. Resulting upstream and downstream wall flow separations also provide regions for radical production and flameholding, but they can also result in locally high wall heat transfer. Compressibility effects noted earlier for mixing layer flows also are evident in the mixing regime downstream of a transverse jet. Compressibility again retards eddy growth and breakup in the mixing layer and suppresses entrainment of fuel and air, resulting in a reduction in mixing and reaction. Noncircular orifice injectors, including elliptical and wedge shaped [59] cross-sections, produce a weaker bow shock and reduced separations, resulting in lower losses and wall heating problems. The lateral spread of the fuel jet is also enhanced, and overall mixing is improved, although there is some reduction in transverse penetration.

Improved mixing has also been achieved using alternative wall injector designs. Wall injection using geometrical shapes that introduce axial vorticity into the flow field has been successful. Vorticity can be induced into the fuel stream using convoluted surfaces or small tabs at the exit of the fuel injector. Alternatively, vorticity can be introduced into the air upstream of the injector using wedge shaped bodies placed on the combustor walls. When strong pressure gradients are present in the flowfield, e.g. at a shock, vorticity aligned with the flow can be induced at a fuel-air interface, where a strong density gradient exists, by virtue of the baroclinic torque. Fuel injection ramps have proven to be an effective means for fuel injection in a scramjet engine. Two ramp injector schemes are diagrammed in Figures 10 a & b. Fuel is injected from the base of the ramp. The unswept ramp configuration provides nearly streamwise injection of fuel to produce a thrust component. Flow separation at the base of the ramp provides a region for flame holding and flame stabilization through the buildup of a radical pool. The ramp itself produces streamwise vorticity as the air stream sheds off of its edges, improving the downstream mixing. The swept ramp design provides all of the features of the unswept ramp, but the sweep results in better axial vorticity generation and mixing. A novel variation on the swept wedge injector, termed the aero-ramp injector, is also shown in Figure 10c. It utilizes three arrays of injector nozzles at various inclination and yaw angles to approximate the physical swept ramp design. The aero-ramp injector has many of the features of the swept ramp design without the losses associated with an intrusive device. A comparison of the two injectors is given in

Figure 11, where transverse fuel penetration, lateral spread, plume area, and mass fraction decay are shown. While transverse penetration and plume area are reduced with the aero-ramp, lateral spread and mass fraction decay are nearly the same as those for the swept ramp injector.

In-stream injection also has been utilized for fuel injection in a scramjet. Traditional approaches involve fuel injection from the sides and the base of an instream strut. Transverse injection results in behavior identical to transverse fueling from the wall. Injection from the base of the strut results in slower mixing as compared to transverse injection. A combination of transverse and streamwise injection, varied over the flight Mach number range, often has been utilized to control reaction and heat release in a scramjet combustor. As noted earlier, however, streamwise injection has the advantage of adding to the thrust component of the engine. To increase the mixing from streamwise injectors, many of the approaches utilized to improve wall injection, including noncircular orifices, tabs, and ramps, have been utilized successfully. Several new concepts have emerged as well. Pulsed injection using either mechanical devices or fluidic oscillation techniques have shown promise for improved mixing. A fluidic approach using a Hartmann-Sprenger tube, shown schematically in Figure 12a, offers a possible means of producing a rapid pressure oscillation with large amplitude by means of a geometrically simple device. Fuel injection schemes integrated with cavities also provide the potential for improved mixing and flameholding. One possible design is shown in Figure 12b. This integrated fuel injection/flameholding device, utilizing fuel injection into a cavity and from its base, integrates the fuel injection with a cavity that provides flameholding, flame stabilization, and mixing enhancement if the cavity is properly tuned.

Even with these results regarding the behavior of mixing layer flows and a number of techniques for enhancing fuel-air mixing, a number of issues remain to be studied. Indeed, a controversy still exists that questions whether fuel-air mixing will even be a problem in a scramjet engine in flight. The issue is whether or not the turbulence present in the atmosphere and ultimately present in the inlet flow will provide sufficient turbulent mixing of fuel and air in the combustor. Since all of the work to study high-speed mixing flows has been conducted (or simulated) using a different (earth bound) environment, the need for enhanced mixing still remains unresolved.

Concluding Remarks

The presentations and discussion periods of the workshop resulted in a number of interchanges between engine developers and members of the associated research community that provided a better understanding of the efforts in each topical area, in keeping with the workshop objectives. The status of the overall engineering effort was described, as were critical needs for successful extensions.

The status of current research in supersonic mixing and combustion was described to the engineering community. A number of plans for future research were discussed. In many instances, the current work and future research plans were consistent with the engineering needs. In other instances, however, needs became apparent that are not being addressed directly.

While the procedures for engine design are well established and fruitful, the inability to make the all of the necessary measurements clearly necessitates further work to develop additional measurement and diagnostic tools. Measurement sensitivities for several critical engine parameters are given in Table 5. Weaknesses requiring improved measurement techniques and devices are also pointed out in the table. Accurate in-stream measurements of velocity, temperature, pressure, and chemical species in engine flow fields using nonintrusive diagnostics are also critical to develop a successful engine design. A summary of diagnostic capabilities for laser-based instrumentation applied to scramjet testing is given in Table 6. The problems for each approach described in the Table must be resolved, or new methods must be developed where necessary. Future approaches must be based on inexpensive and robust technologies, and the resulting instrumentation must be useful in hostile testing environments.

Simulation and modeling capabilities must be extended to allow more routine application to realistic engine geometries. An order of magnitude increase in computational speed must be achieved before engine design codes can meet this challenge. Multigrid methods appear to be one approach for achieving this goal, but significant work is needed before applying this method to high-speed reacting flows. Improving computer architectures, particularly parallel processors, also will provide some of the needed enhancement. Turbulence modeling also requires significant work. Research is needed not only to improve the capability for modeling the flowfield turbulence, but also to describe the interaction of turbulence with chemistry in a compressible reacting flow. For the analysis of engine component flows, large eddy simulation may provide a means for computing (rather than modeling) a larger proportion of the scale of the flow. To model chemical reaction of fuel and air in an engine, reduced kinetic models must be developed to reduce computational time required for solving the species equations, particularly for hydrocarbon fuels. To support hydrocarbon-based scramjet engine development, a comprehensive data base for C_7 - C_{12} aliphatic fuel components under scramjet conditions should be developed. In addition, surrogate hydrocarbon fuels should be selected based upon available information about endothermic behavior and catalysis. And finally, the aliphatic fuels data base should be utilized to derive suitable starting and reduced mechanisms for candidate fuels.

Several approaches for fuel injection in the combustor were discussed. Designs utilizing geometries or flow alterations that induce streamwise vorticity to enhance mixing appear to be most promising. Losses induced by the injection process reduce the efficiency of the injector in most designs, however. Future work to optimize the injector design for maximum mixing enhancement with minimum losses will be needed. Work to relate findings from simulations or ground based testing to actual conditions in flight should be included.

Issues regarding the thermochemical and transport behavior of the fuels were also raised. A better understanding of the state of hydrocarbon fuels as their temperature increases in vehicle cooling passages is important for design. Changes in the state of the fuel can affect the reactivity of the fuel and the resulting combustion efficiency significantly. There is a lack of understanding of the physical processes that may contribute to these effects. To understand these phenomena, changes in the fuel state must be studied in a realistic simulation of the scramjet preheating and combustion processes. A co-annular or opposed-jet burner that would burn hot, partially reacted fuels represents one possible relevant experiment for such studies. Traditional premixed combustion devices appear to be inadequate to address the important issues.

Acknowledgment

The authors wish to thank all of the participants in the workshop for volunteering to attend and for their valuable contributions to the discussions. The authors especially express their gratitude to the presenters at the workshop, not only for the presentations, but also for their comments to improve our attempts to summarize their thoughts in this paper.

References

1. Billig, F. S. "Propulsion Systems From Takeoff to High Speed Flight," Reprint from High-speed Flight Propulsion System, Vol. 137 of Progress in Astronautics and Aeronautics, AIAA.

2. Kay, I. W. et al., "Hydrocarbon Fueled Scramjet, Vol. VIII, Piloting and Flame Propagation Investigation," AFAPL TR 68-146, May 1971.

3. Heins, A. E., et al., "Hydrocarbon Scramjet Feasibility Program Part III. Free Jet Engine Design and Performance," AFAPL TR-74, Jan 1971.

4. Billig, F. S., "Combustion Processes in Supersonic Flow," Journal of Propulsion and Power, Vol. 4, May-June 1988, pp. 209-216.

5. Burnette, T. D., "Dual Mode Scramjet," The Marquardt Corporation, AFAPL-TR-67-132, June 1968.

6. Harshman, D. L., "Design and Test of a Mach 7 - 8 Supersonic Combustion Ramjet Engine," AIAA Propulsion Specialist Meeting, July 1967, Washington, DC.

7. Billig, F. S., "Research in Supersonic Combustion," AIAA Paper 92-0001, Jan 1992

8. Anderson, G. Kumar, A., and Erdos, J. "Progress in Hypersonic Combustion Technology with Computastion and Experiment," AIAA Paper 90-5254, Oct. 1990.

9. Voland, R. and Rock, K., "Concept Demostration Engine and Subscale Parametric Engine Tests," AIAA Paper 95-6055, April 1995.

10. Stevens, R. Thompson, M. W., "Steam Calorimetry Uncertainty Analysis for Direct-Connect Arc Tunnel Combustor tests, GWP # 107A," APL-NASP-94-016, Jan. 1995

11. JANNAF Ramjet Subcommittee Workshop, "Fuels for Future High-Speed Flight Vehicles," CPIA Publication 518 (2 vol.), June 1988.

12. Northam, G. B., ed., "Workshop Report: Combustion in Supersonic Flow," 22nd JANNAF Combustion Meeting Proceedings, CPIA Publication 432, Vol. 1, 1985.

13. Beach, H. L., "Supersonic Combustion Status and Issues," in Major Research Topics in Combustion, Hussaini, Kumar, and Voigt, eds, Springer Verlag, NY, 1992.

14. Heiser, W. H., Pratt, D. T., Hypersonic Airbreathing Propulsion, AIAA.

15. Murthy, S. N. B., and Curran, E. T., eds, High-Speed Flight Propulsion Systems, AIAA Progress in Astronautics and Aeronautics, Vol. 137, AIAA, 1991.

16. Sobel, D. R., and Spadaccini, L. J., "Hydrocarbon Fuel Cooling Technologies for Advanced Propulsion," ASME-95-GT-226, June 1995.

17. New World Vistas, Air and Space Power for the 21st Century, Aircraft and Propulsion Volume, 1996.

18. Ianovski, L. S., "Endothermic Fuels for Hypersonic Aviation," Symposium on Fuels and Combustion Technology for Advanced Aircraft Engines, AGARD-CP-536, pp. 44-1 to 44-8, Sept. 1993.

19. Ianovski, L., Sapgir, G., "Heat and Mass Transfer to Hydrocarbon Fuels at Thermal Decomposition in Channels of Engines," AIAA Paper 96-2683, July, 1996.

20. Lander, H., Nixon, A. C., "Endothermic Fuels for Hypersonic Vehicles," Journal of Aircraft, Vol. 8, No. 4, pp. 200-207, 1971.

21. Favorskii, O. N., Kurziner, R. I., "Development of Air-Breathing Engines for High Speed Aviation by Combining Advances in Various areas of Science and Engineering," High Temperature (translation of Teplofizika Vysokikh Temperature), Vol. 28, No. 4, pp. 606-614, 1990.

22. Warnatz, J., "Rate Coefficients in the C/H/O System," in Combustion Chemistry, W. C. Gardiner, ed., Springer Verlag, New York, pp. 197-360, 1984.

23. Spadaccini, L. J., Colket, M. B., "Ignition Delay Characteristics of Methane Fuels," Prog. Energy Combust. Sci., Vol. 20, pp. 431-460, 1994.

24. Faith, L. E., Ackerman, G. H., Heck, C. K., Henderson, H. T., Ritchie, A. W. Ryland, L. B., "Hydrocarbon Fuels for Advanced Systems," AFAPL-TR-70-71, Dec. 1971. (AIAA Journal, Vol. 4, pp. 513-520, 1966).

25. Edwards, T., "Recent Research Results in Advanced Fuels," ACS Petroleum Chemistry Division Preprints, Vol. 41, No. 2, pp. 481-487, 1996.

26. Savage, P. E., et al, "Reactions at Supercritical Conditions: Applications and Fundamentals," AIChE Journal, Vol. 41(7), pp. 1723-1778, 1995.

27. Dardas, Z., et al, "High-Temperature, High-Pressure in Situ Reaction Monitoring of Heterogeneous Catalytic Processes under Supercritical Conditions by CIR-FTIR," J. Catalysis, Vol. 159, pp. 204-211, 1996.

28. Pellett, G. L., Northam, G. B., Wilson, L. G., "Counterflow Diffusion Flames of Hydrogen, and Hydrogen Plus Methane, Ethylene, Propane, and Silane, vs Air: Strain Rates at Extinction," AIAA Paper 91-0370, Jan. 1991.

29. Edwards, T., "USAF Supercritical Hydrocarbon Fuels Interests", AIAA Paper 93-0807, January 1993.

30. Chen, L.-D., Sui, P.-C., "Atomization During the Injection of Supercritical Fluid Into High Pressure Environment," paper presented at 1994 IUTAM on Droplets and Sprays, 6-10 Dec. 1994, Tainan, Taiwan.

Publishers, SA, 1996.

31. Petley, D. H., Jones, S. C., "Thermal Management for a Mach 5 Cruise Aircraft Using Endothermic Fuel," Journal of Aircraft, Vol. 29(3), pp. 384-389, 1992.

32. Pegg, R. J., Hunt, J. L., Petley, D. H., "Design of a Hypersonic Waverider-Derived Airplane," AIAA Paper 93-0401, 1993.

33. Egolfopoulos, F. N. and Law, C.K., "Chain mechanisms in the Overall Reaction Orders in Laminar Flame Propagation," Combustion and Flame, Volume, 80, pp. 7-16, 1990.

34. Peters, N., "Numerical and Asymptotic Analysis of Systematically Reduced Reaction Schemes for Hydrocarbon Flames," Numerical Simulation of Combustion Phenomena, Lecture Notes in Physics, Vol. 241, pp. 90-109, 1985.

35. Lam, S. H., "Reduced Chemistry Modeling and Sensitivity Analysis," 1994-1995 Lecture Series: Aerothermochemistry for Hypersonic Technology, Von Karman Institute for Fluid Dynamics, 1995.

36. Chelliah, H. et al., "Reduced Kinetic Mechanisms for Application in Combustion Systems," Peters and Rogg, editors, Springer Verlad, 1993.

37. Pope, S. B., "Computationally Efficient Implementation of Combustion Chemistry Using In-Situ Adaptive Tabulation," Cornell University Report FDA 96-02, 1996.

38. Eckbreth, A.C., Laser Diagnostics for Combustion Temperature and Species, Second Edition. Combustion science and Technology Book Series, Vol. 3, Gordon and Breach Publishers, SA, 1996.

39. Hanson, R. K., "Combustion Diagnostics: Planar Flowfield Imaging," Twenty First Symposium (International) on Combustion, The Combustion Institute, pp. 1677-1691, 1986.

40. Edwards, J. R., "Development of an Upwind Relaxation Multigrid Method for Computing Three-Dimensional Viscous Internal Flows," AIAA Paper 95-0208, Jan. 1995.

41. Adumitroaie, V., Colucci, P. J., Taulbee, D. B., and Givi, P., "LES, DNS and RANS for the Analysis of High-Speed Turbulent Reacting Flows," Annual Report, NASA Grant NAG 1-1122, 1994.

42. Girimaji, S. S., Galilean, A., "Invariant Explicit Algebraic Reynolds Stress Model for Curved Flows," NASA CR-198340, pp. 1-28 June 1996.

43. Colucci, P. J., Jaberi, F. A., Givi, P., and Pope, S. B., "Filtered Density Function for Large Eddy Simulation of Turbulent Reacting Flows," Submitted for publication, Physics of Fluids.

44. Jachimoskski, C. J., "An Analytical Study of the Hydrogen-Air Reaction Mechanism with Application to Scramjet Combustion," NASA TP-2791, Feb. 1988.

45. Eklund, D. R., Northam, G. B., and Hartfield, R. J., "A Detailed Investigation of Staged Norman Injection into a Mach 2 Flow," Proceedings of the 27th JANNAF Combustion Meeting, Cheyenne, Wyoming, Nov. 5-9, 1990.

46. Delarue, B. J., and Pope, S. B.,

"Application of PDF Methods to Compressible Turbulent Flows," Submitted for publication, Physics of Fluids.

47. Mass, U., and Pope, S. B., "Simplifying Chemical Kinetics: Intrinsic Low-Dimensional Manifolds in Composition Space," Combustion and Flame, V. 88, pp. 239-264, 1992.

48. Dimotakis, P. E., "Turbulent Free Shear Layer Mixing and Combustion," High Speed Flight Propulsion Systems, Chapter 7, Progress in Astronautics and Aeronautics, vol. 137, 1991.

49. Drummond, J. P., Carpenter, M. H., and Riggins, D. W., "Mixing and Mixing Enhancement in High Speed Reacting Flows, High Speed Flight Propulsion Systems," Chapter 7, Progress in Astronautics and Aeronautics, vol. 137, 1991.

50. Givi, P., and Riley, J. .J., "Some Current Issues in the Analysis of Reacting Shear Layers: Computational Challenges," Major Research Topics in Combustion, Editors: M. Y. Hussaini, A. Kumar and R. G. Voigt, pp. 588-650, Springer-Verlag, New York, NY,
1992.

51. Drummond, J. P., and Givi, P., "Suppression and Enhancement of Mixing in High-Speed Reacting Flow Fields," Combustion in High-Speed Flows, pp. 191-229, Editors: J. Buckmaster, T.L. Jackson and A. Kumar, Kluwer Academic Publishers, Boston, MA, 1994.

52. Goebel, S. G., and Dutton, J. C., "velocity Measurements of Compressible, Turbulent Mixing Layers," AIAA Paper 90-0709, January 1990.

53. Smith, S. H., Lozano, A., Mungal, M. G., and Hanson, R. K., "Scalar Mixing in the Subsonic Jet in Crossflow." AGARD Computational and Experimental Assessment of Jets in Crossflow, Winchester, pp. 6.1-6.13, 1993.

54. Hall, J. L., Dimotakis, P. E., and Rosemann, H. "Experiments in Non-reacting Compressible Shear Layers," AIAA Paper 91-0629, January 1991.

55. Samimy, M., Zaman, K. B. M .Q., and Reeder, M. F., "Effect of tabs on the Flow and Noise Field of an Axisymmetric jet," AIAA Journal, Vol. 31, No. 4, pp. 609-619, 1993.

56. Cox, S. K., Fuller, R. P., Schetz, J. A., and Walters, R. W., "Vortical interactions Generated by an Injector Array to Enhance Mixing in Supersonic Flow" AIAA Paper 94-0708, January 1994.

Appendix A

Research Problems for Future Work

- Affordability - minimize weight, size,complexity, part count for a given mission profile
- Methodology for optimization
- Materials and structures
  - scaling of leading edges to minimize drag
  - "cheating" by injecting liquid, ablating, etc.
- Inlet design
- Fuel characterization
- Unsteadiness
- Fuel injection and mixing

- Recovery of kinetic energy to produce thrust

GROUND BASED TESTING

- Influence of contaminants on ignition-vitiation; combustion and expansion besides ignition; effects on radiation
- Turbulence - no current data
- Nonequilibrium
- Boundary layer effects
- Scaling to flight conditions
- Complementary experimental program for relevant phenomena
- Utilization of pulsed facilities
  - elimination of vitiation effects

FUELS AND FUEL SYSTEMS

- Experimental program to determine energy yields of fuels
- Creation of kinetics data base for long-term use
- Low temperature starting and piloting systems
  - trimethyl Al additives
  - GASL micro rocket
  - plasma torch
  - embedded ramjets
  - gelled fuels (GASL)
- Improvement of fuel specific ...
- Nano particle carbon particles
- Micro encapsulated fuels

INJECTION AND MIXING

- Exploitation of longitudinal vorticity for mixing enhancement
- Interaction between injectors
- Minimization of losses
- Thermodynamic state of fuel at injection
- Cold flow studies?
- Curvature-induced Rayleigh destabilization; role of pre-existing turbulence
- Systems studies to optimize, but not necessarily minimize, losses
- German-Russian work on three injector classes-micro pylons
- Relationship to flame holding

COMBUSTION CHEMISTRY

- Compile and validate kinetic data base at three levels
  - detailed
  - skeletal
  - reduced
- Ignition enhancers
- Liquid-phase kinetics; supercritical kinetics
- Recombination kinetics
  - catalytic additives
- Role of soot
- Combustion at high strain rates
- Unsteadiness
- Incorporating kinetics mechanisms in design codes
- Development of subscale experiments
  - Russian results by Baev?
  - opposed jet burner?

DIAGNOSTICS

- Skin friction measurements
- Heat flux measurements, including radiation
- Detailed measurements of boundary and initial conditions
- Application of non-intrusive instrumentation to free jet tests
- Measurement of velocity profiles
- Determination of measurement uncertainties

- Turbulence intensity levels (concentration in supersonic flow)
- Instantaneous measurements to determine turbulence-chemistry interactions
- Pressure-sensitive paint to measure surface pressures
- Mapping of total pressure and total temperature
- Design of well-posed experiment
- Concentration measurements
  - mean and fluctuating
  - spectrally resolved

SIMULATION

- Stochastic models
- Sensitivity to unsteadiness
- Algebraic closure models
  - stress
  - scalar flux
- Solvers for particle methods
  - improved efficiency
- Well-posed validation experiment
- Preprocessing
  - adaptive gridding
- Solvers
  - increased efficiency (factor of 10)
  - provisions for real time (dynamic) grid adaptation
  - domain and function decomposition capability for massively parallel and/or networked computers
- Physical models
  - turbulence/chemistry interaction models
  - testing of higher order phenomenological turbulence models
  - assessment of LES techniques for realistic geometries and flow conditions
  - testing of fast reduced kinetics mechanisms

BILLIG'S COMMENTS

- Inlets-Isolators
  - streamline tracing
  - analogy between C-I-I & aerodynamic phenomena
  - shear high temperature reduction
  - sweep
  - starting
- Fuels
  - densification
  - additives
  - storability
  - toxicity
  - rheology
- Fuel Preparation
  - heat pipes (open-closed)
  - plasma generators
- Injection - Mixing
  - subsonic imbedded zones (cavities, steps, bases)
- Ignition
  - radical generators
- Combustion-Combustors
  - physical vs. thermal throats
  - shear, high temperature
  - recombination kinetics
  - transpiration
- Nozzles
  - shear, high temperature
  - recombination
  - exploitation of non-uniform entrance flow

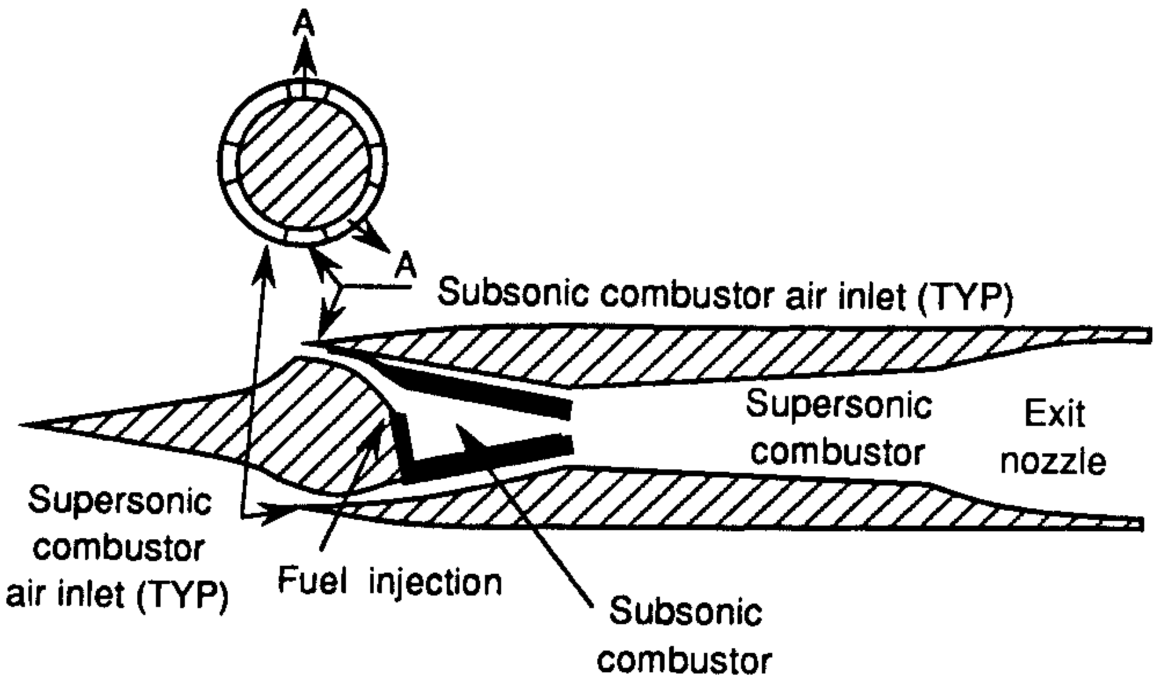


Figure 1 Schematic illustration of Dual Combustor Engine

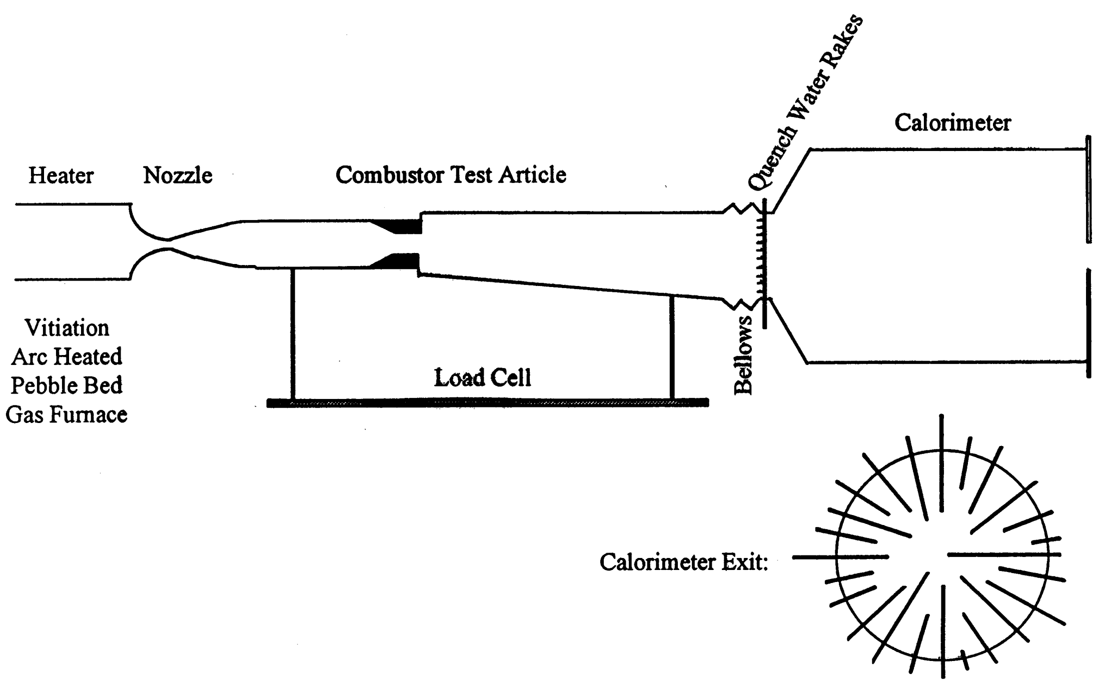


Figure 2. Schematic Illustration of Direct Connect Combustor Test Facility

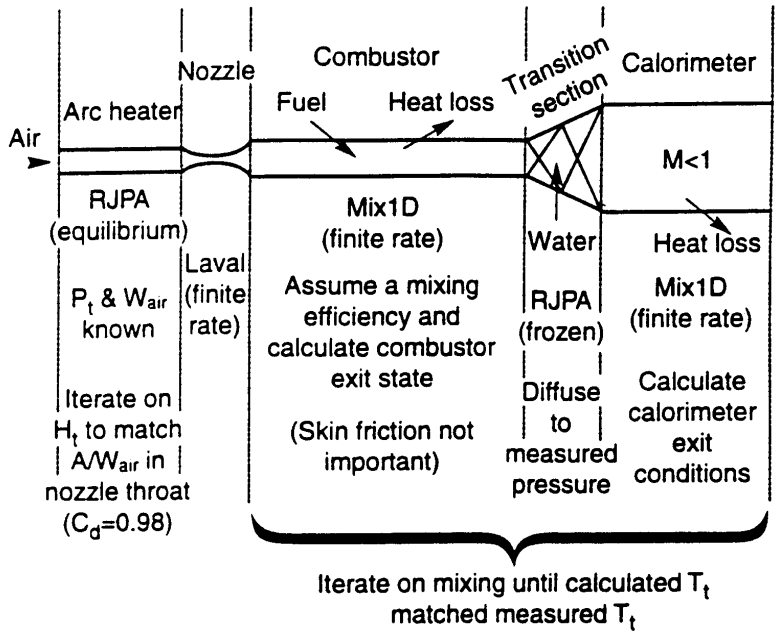


Figure 3. Schematic Illustration of Steam Calorimetry Data Analysis Procedure

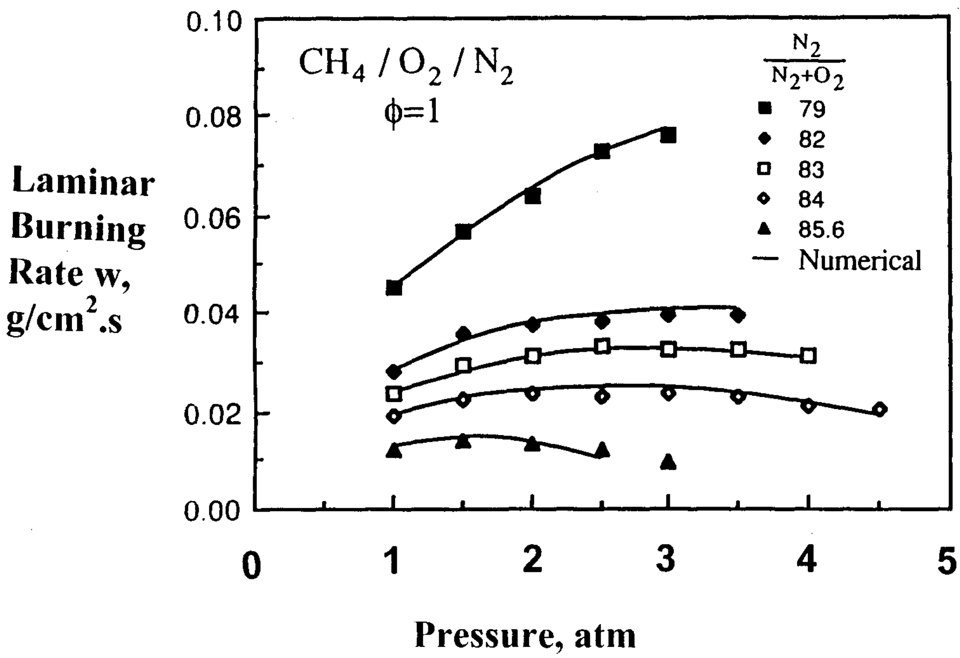


Figure 4. Laminar Burning Rate vs. Pressure

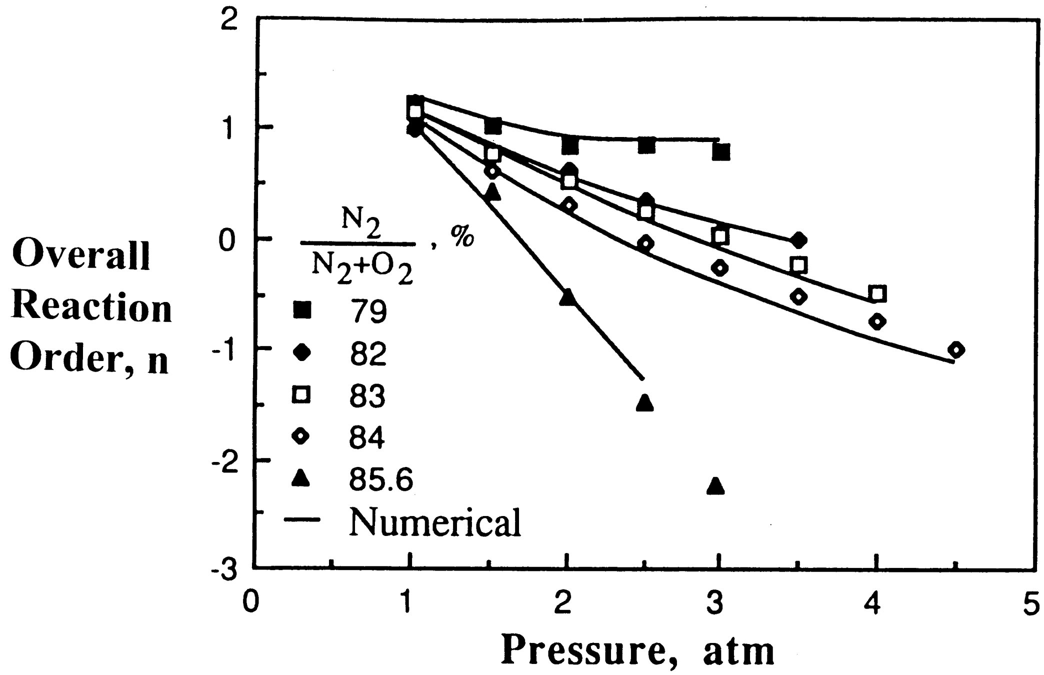


Figure 5. Overall Reaction Order vs. Pressure

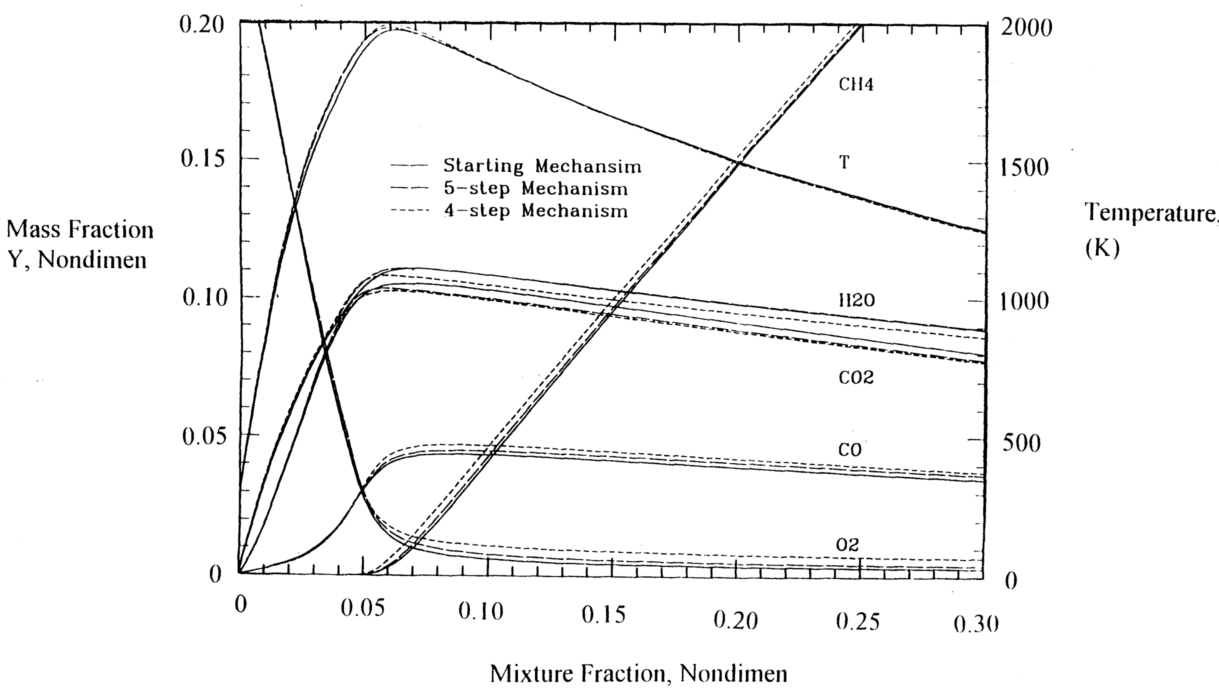
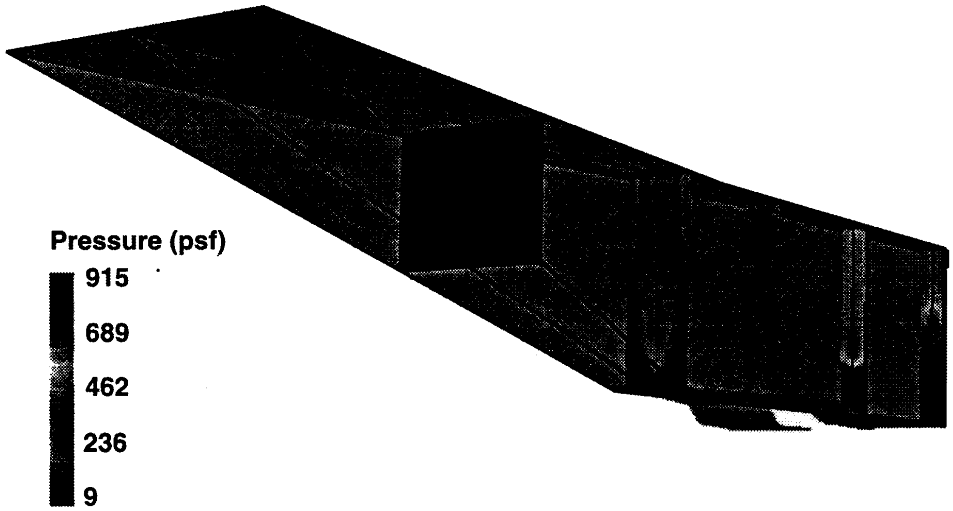


Figure 6. Mass Fraction and Temperature vs. Mixture Fraction



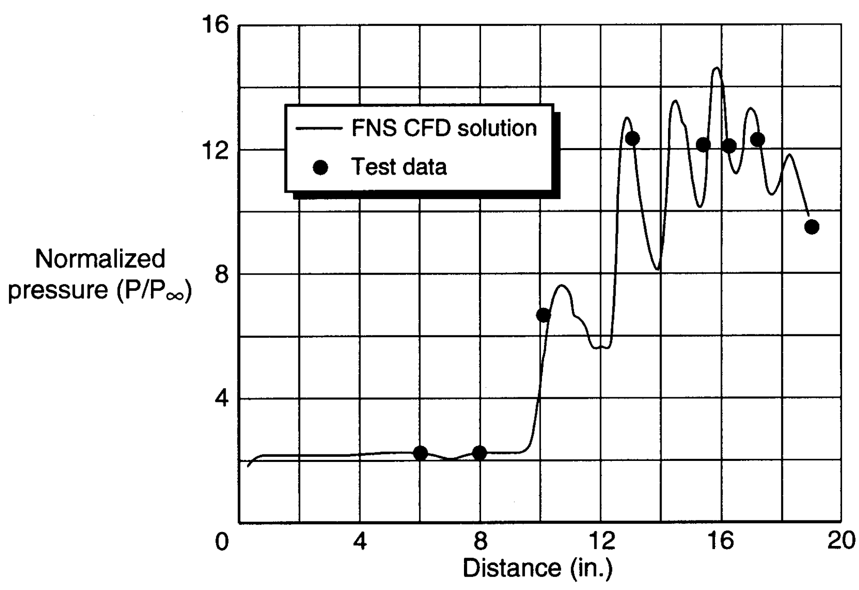
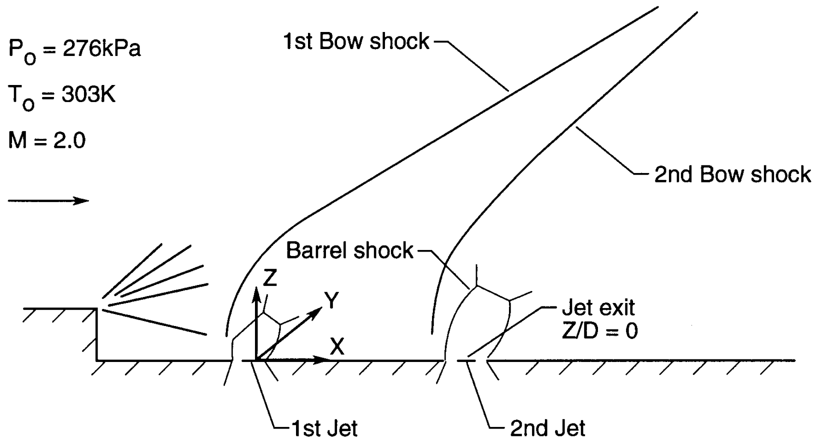


Figure 7. Typical Side Wall Compression Inlet Calculations



Schematic of Tandem Fuel Injection Behind a Step in Supersonic Flow

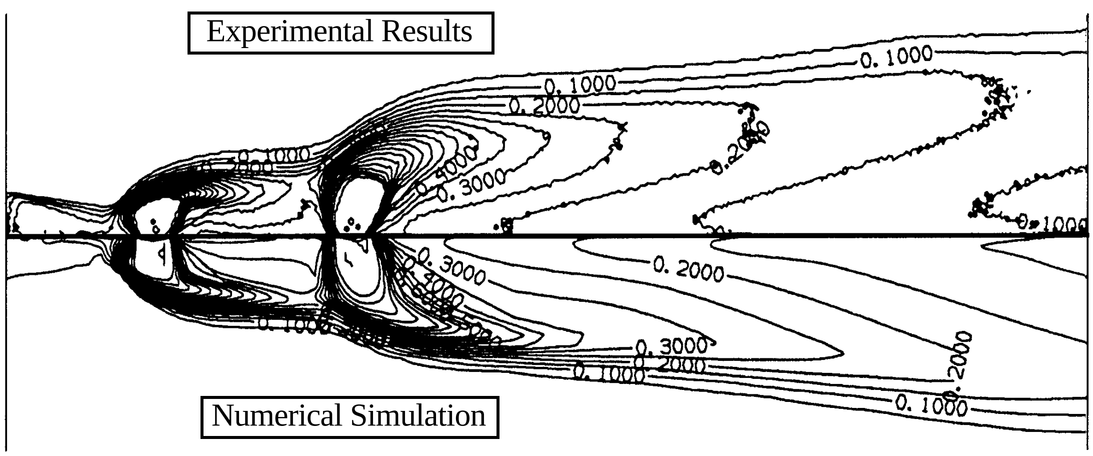


Figure 8. Calculations of the near Field of a Transverse Fuel Injector Configuration

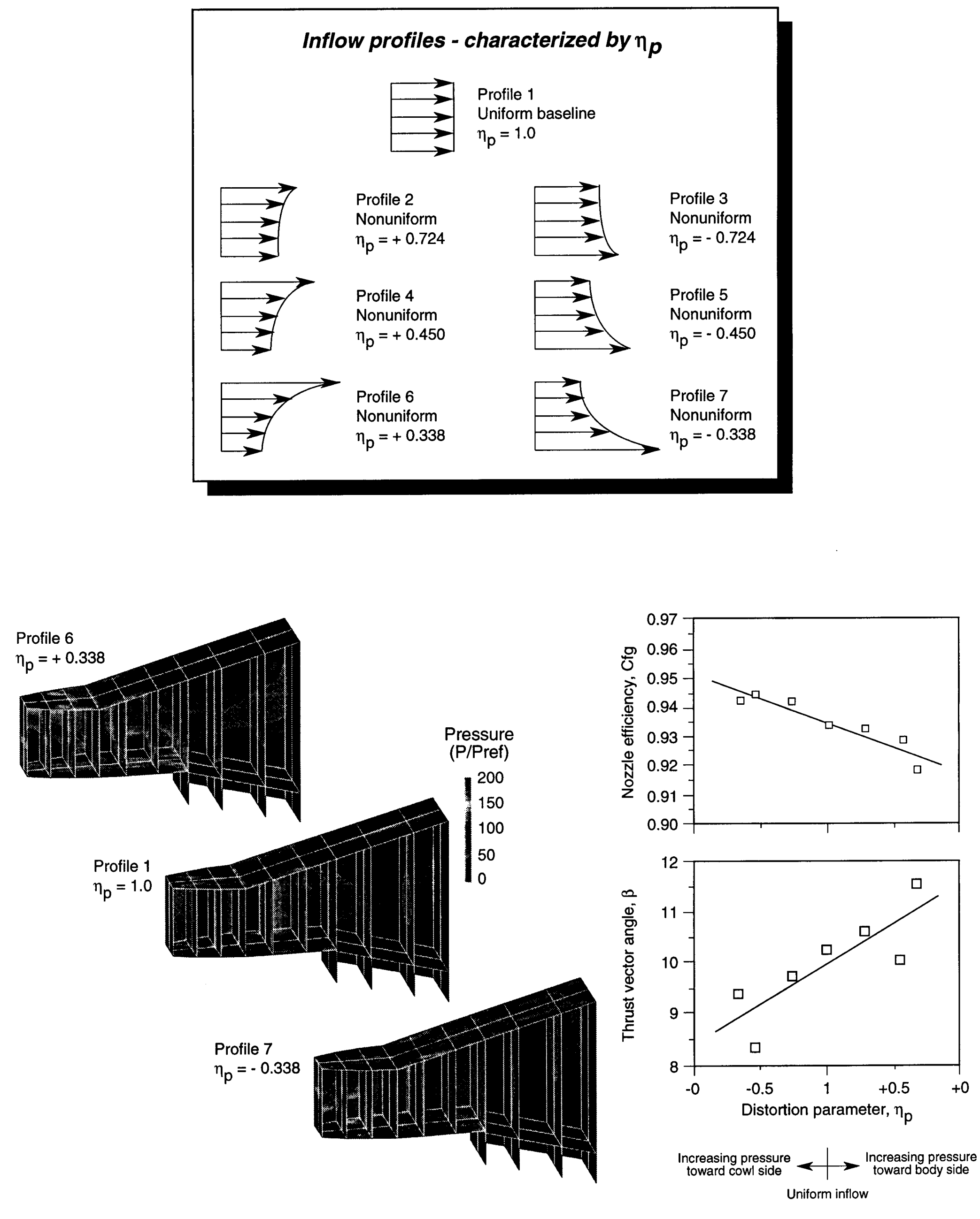


Figure 9. Optimization of Nozzle Performance Based on Nozzle Inflow Profiles

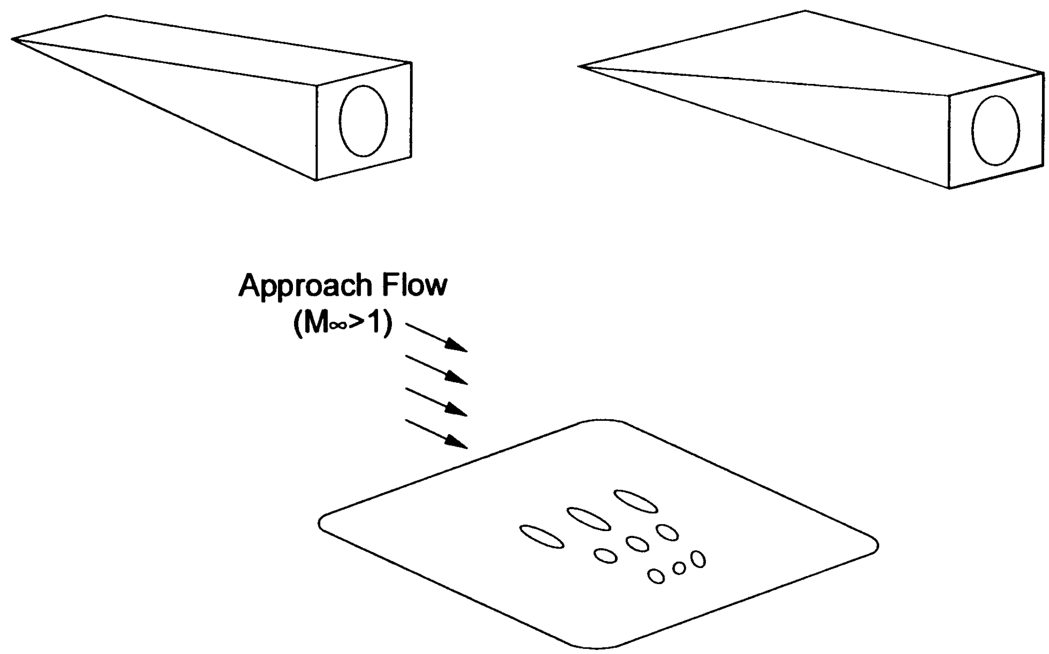


Figure 10 (a, b, c). Schematic Illustration of Ramp Fuel Injectors for Scramjet Engines
a) Unswept, b) Swept, c) Aero-Ramp

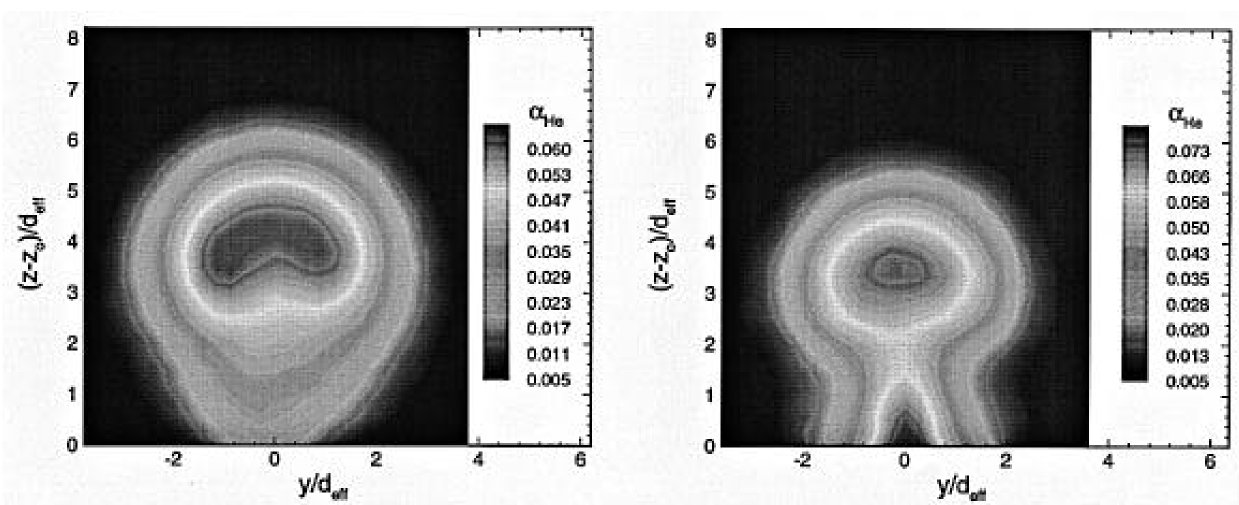
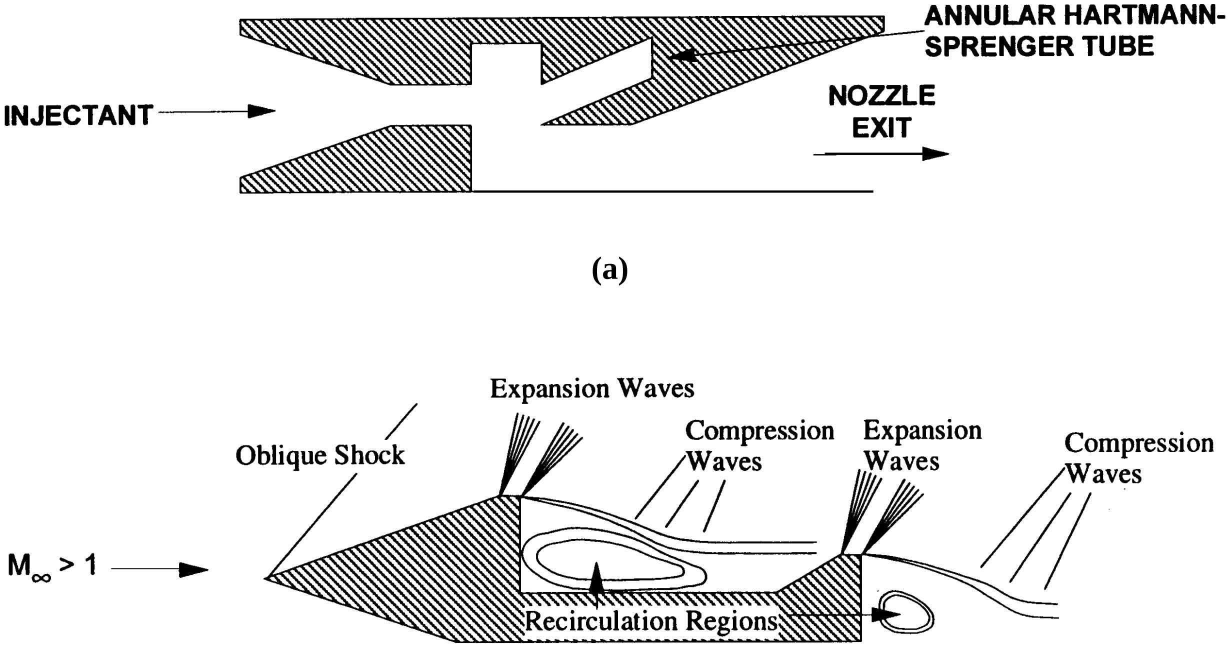


Figure 11. Comparison of Ramp and Aero-Ramp fuel Injectors
(fuel mass fraction contours)



(a)

(b)

Figure 12(a, b). Schematic Illustration of Pulsed and Cavity Injector-Flameholders Concepts. a) Hartmann-Sprenger Tube, b) Integrated Injector-Flameholder